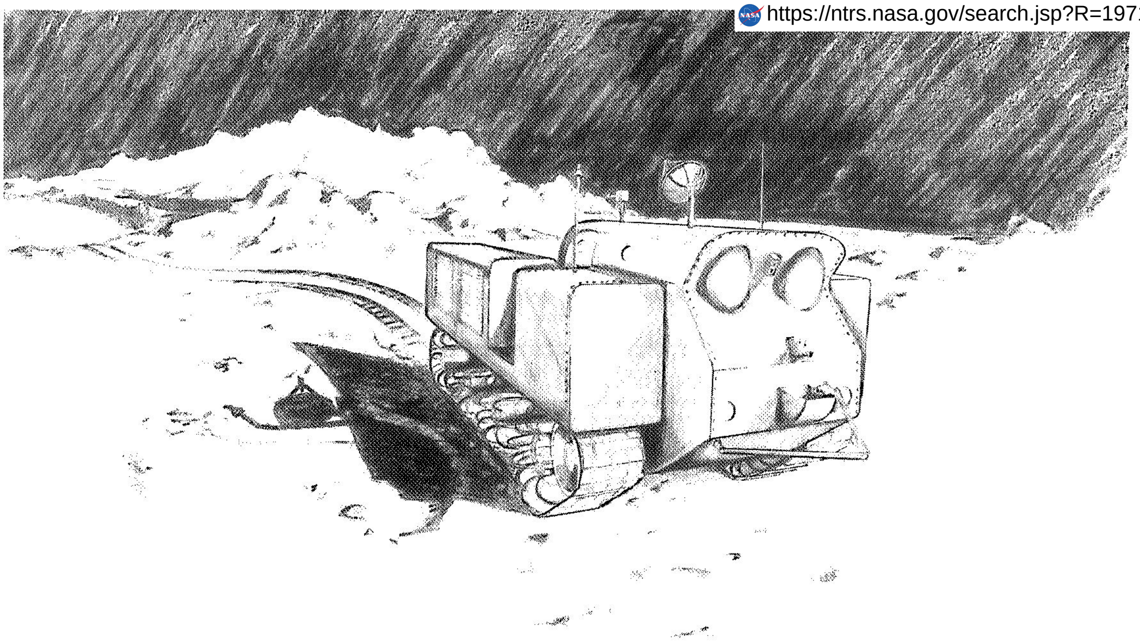

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MULE

Manned-Unmanned Lunar Explorer

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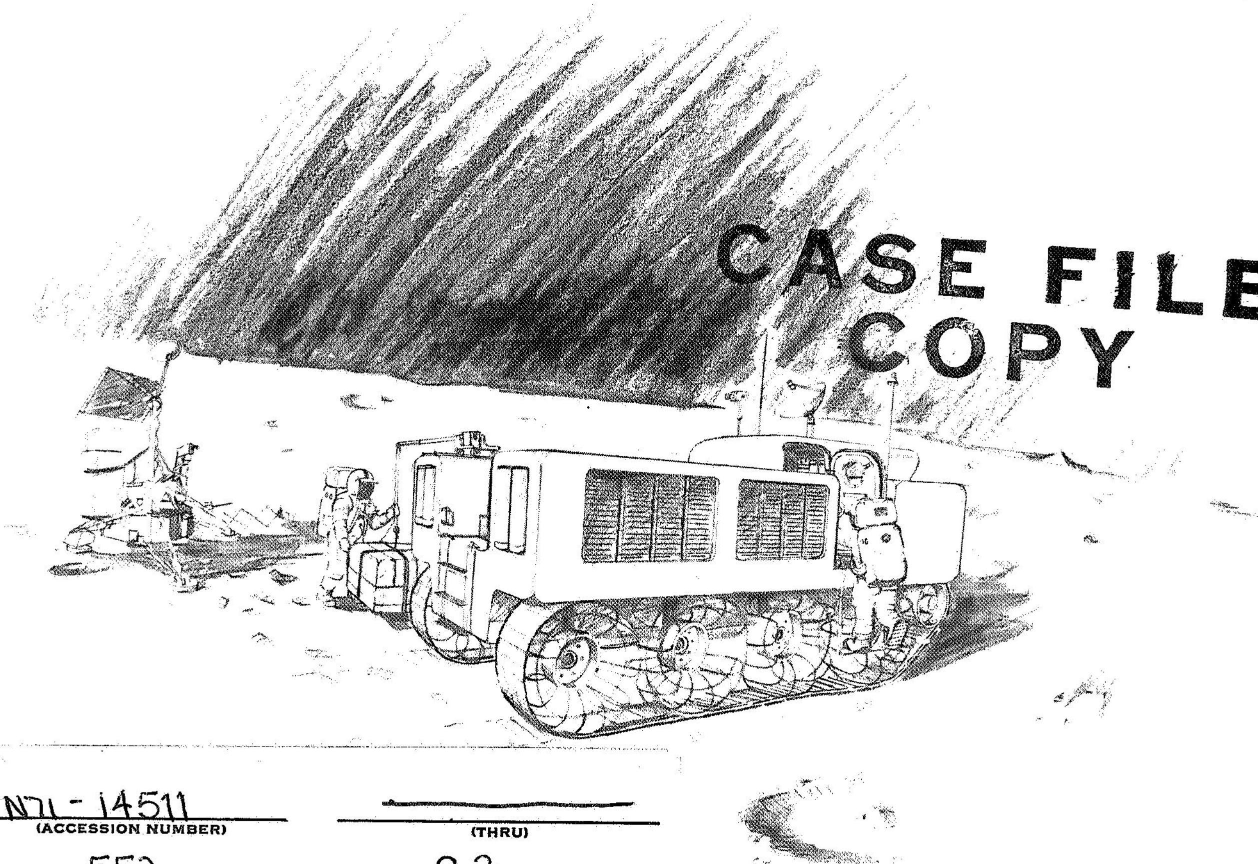

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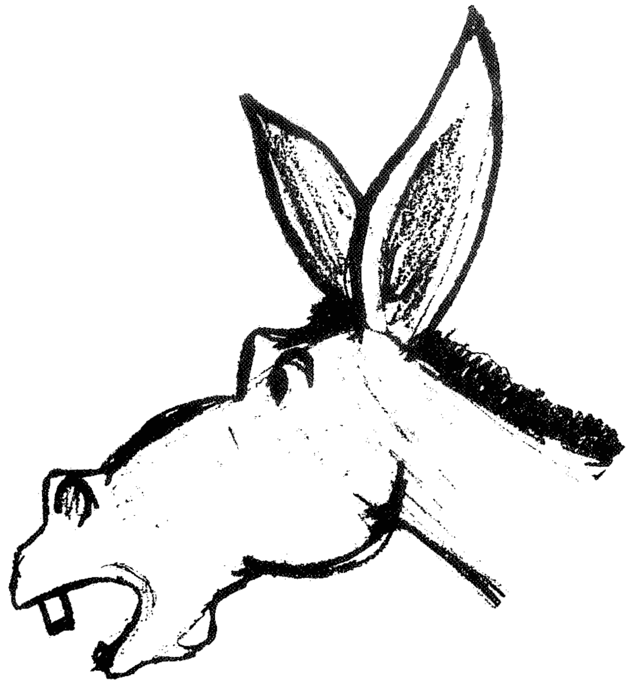

CONCEPTUAL DESIGN

of a

MANNED-UNMANNED LUNAR EXPLORER

September, 1970

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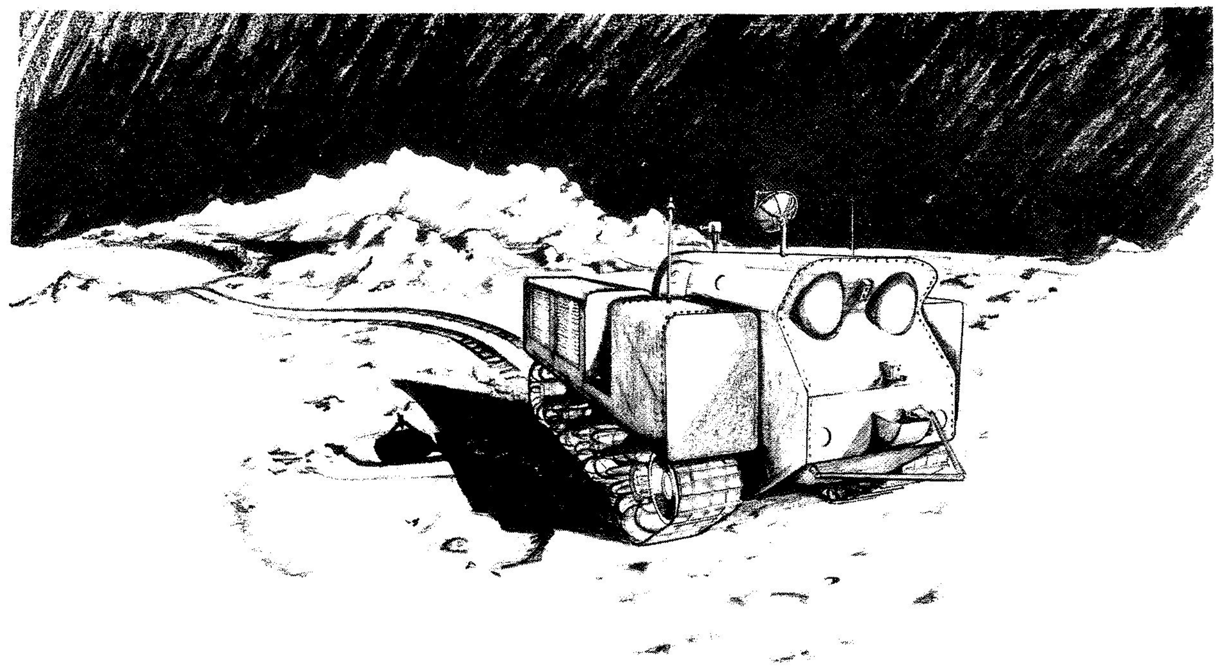

THE MULE

FOREWORD

R. Doyle Holstead

This document reports the activities and results of the 1970 NASA/ASEE Summer Faculty Fellowship Institute for Engineering Systems Design at University of Houston, NASA-Manned Spacecraft Center, and Rice University. This was the fourth consecutive year for the Institute. The broad objectives of the Institute were "to increase competence and to develop concepts which will enable participants to organize multidisciplinary engineering system design programs at their home institutions and to establish and further communication and collaboration between engineering and other disciplines".

An effective means of teaching engineering systems design is to have the student participate in a design project that employs the systems approach with adequate guidance and direction given by those experienced in systems design. In order to meet these objectives by this means of teaching, twenty faculty members of engineering and science from universities and colleges throughout the United States were selected to participate in a systems design project. A list of participants and staff follows.

The design of a mobility aid to augment manned exploration of the lunar surface provided a complex problem whose solution demanded the talents of many engineering and scientific disciplines frameworked within a systems design approach. Thus the selected faculty members became students in engineering systems while producing a design of a mobility aid that may be used by NASA-Manned Spacecraft Center. Other objectives of a more personal quality that were satisfied by the Institute were:

- The participants better understand the NASA's operations, goals, and contributions.
- The participants experienced the interactions of a group and realized that the success of the group depends on communication and teamwork.
- The participants experienced the triumphs and frustrations inherent in project management both as a leader and as a follower within the group.
- The participants realized that there are many contributions to technology still to be made.

1970 NASA-ASEE

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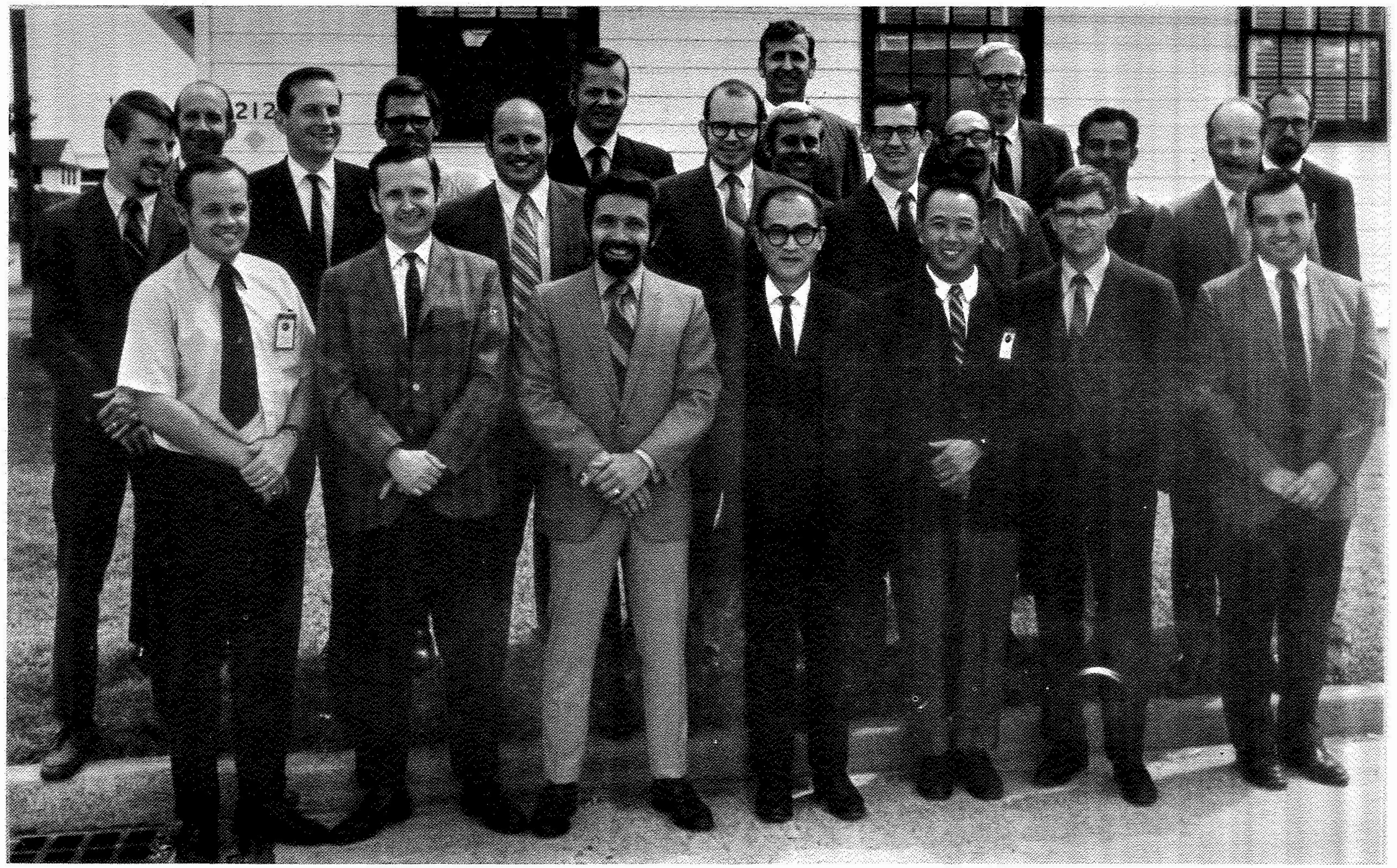

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ACKNOWLEDGMENTS

The participants and staff acknowledges the encouragement and aid given by many persons. Mr. Harry Huntoon and Miss Joan Hagan of the University Affairs Office at MSC provided administrative assistance at all hours. Mr. Bass Redd of MSC detailed Mr. Dave Howes to provide direct technical liaison with the institute. This proved a most fortunate choice.

We wish to thank President P. G. Hoffman and Dean C. V. Kirkpatrick of the University of Houston. Also, we thank Mrs. Inez Law who acted as executive secretary to the Institute, and Mrs. Nel Fowler, Mrs. Lorraine Gallegos, and Mrs. Jesselyn Ealim who did a splended typing job while listening to directions from more than twenty bosses simultaneously.

Finally, we wish to acknowledge the many speakers who provided us with information about the design problem and related topics. A list is given in the following table.

SPEAKERS

Name	Topic	Affiliation
Robert Abel	Lunar Visibility	MSC
Dick Bailey	Lunar Exploration PDD	MSC
Robert Bond	Surface Transportation PDD	MSC
Bill Davis	Space Programs for Decade of '70's	MSC
Jerry Davis	Television Technology	MSC
Steward Davis	Chemical Propulsion Stage PDD and Space Tug	MSC
Steve Dickerson	System Design	Georgia Tech.
Ralph Everett	Computers at MSC	MSC
Burt Fraser	35mm Slides as Visual Aids	U of H
Jerry Goodman	Equipment Related Human Factors	MSC
John Jackson	Human Performance in Space	MSC
Roger Koppa	Astronaut Training for Lunar Surface Operations	General Electric
John Lee	Space Station	MSC
Gary McBride	Time & Motion Analysis for Astronaut Lunar Surface Activities	Fordham University
R. L. Motard	SERV (1967)	U of H
Emanuel Schnitzner	Teleoperators and Remote Control	MSC
Retha Shirley	MSC Library	MSC
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ABSTRACT

The Manned, Unmanned Lunar Explorer, MULE, is a conceptual system design by the 1970 NASA-ASEE Summer Faculty Fellowship Institute for Engineering System Design at the University of Houston, NASA-Manned Spacecraft Center, and Rice University. The participants were twenty faculty members of engineering and science from nineteen institutions within the United States and were aided by a staff of six additional faculty members. The Institute was held during the eleven week period from June 8 thru August 21.

The objectives of the Institute were twofold: to develop competence and concepts which will enable the participants to organize system design programs at their institutions; and to produce a conceptual design of a mobility aid to augment exploration of the lunar surface. These objectives were accomplished simultaneously; the MULE was designed by using the methodology of formal engineering systems design. This report is devoted to the design of the MULE with the methodology of systems design implied.

The MULE was conceived to be operational in the 1980's and to be compatible with the Integrated Program Plan for the national space effort. The MULE has a gross weight on earth of 9705 pounds and mission capabilities of 36 hours, 250 kilometers in the manned mode and 1500 kilometers in the unmanned mode. It employs a two-man crew, uses tracks for locomotion, and transports a science payload of 2000 pound.

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CHAPTER 1

INTRODUCTION

R. Doyle Holstead

The objectives of the 1970 NASA/ASEE Summer Faculty Fellowship Institute for Engineering Systems Design at the University of Houston, NASA-Manned Spacecraft Center, and Rice University were twofold: to develop competence and concepts which will enable the participants to organize system design programs at their institutions; and to produce a conceptual design of a lunar mobility aid for the NASA-Manned Spacecraft Center. The Institute was directed such that these objectives were accomplished simultaneously; the mobility aid was designed using formal engineering systems design methods. Thus engineering system design was learned mainly by participation in an actual design project. This report will not discuss explicitly the methods of engineering systems design, but these methods will be implied. This report will be devoted to the development of the conceptual design of the lunar mobility aid.

The conceptual design of a lunar mobility aid was provided as an objective by the Manned Spacecraft Center by means of a preliminary "Statement of Work". This "Statement of Work" was the predominant guiding factor for the work of the Institute, and since it sets forth the problem, it is presented in its entirety in Appendix A. The "Statement of Work" divided the design into two phases. Chapter 2 presents a chronological and cursory description of the work performed in Phase I and Phase II. The other sections of the report present the details of the work performed by the individuals and the groups of the Institute.

Part I presents the requirements established for the mobility aid, and the methods used to establish these requirements. Part II presents the methods employed to synthesize the mobility aid and Part III describes the resulting mobility system and subsystems. Part IV discusses the requirements that must be met to support the development and utilization of the system.

The major milestones reached by the Institute are presented in Table 1-1.

Table 1-1

Institute Time Schedule

June 8	Institute began
June 16	Team organized; group leaders elected
June 22	Proposal presented to MSC
June 24	Phase I began
July 16	Phase I ended; presented to MSC
July 17	Phase II began
July 20	Team reorganized; group leaders elected
August 18	Phase II presented to MSC
August 21	Contributions to final report submitted; program ended.

CHAPTER 2

SUMMARY

A. J. Perna

This section contains a summary of the events, methods and procedures used in the Summer Design Institute Study. It is developed and presented in the chronological fashion that the study evolved in order to establish the logical sequence followed in analyzing and solving the problem posed. Only brief or cursory descriptions relating to the areas of study are presented in the main body under their respective headings.

The design team was formulated with the intent of rendering and developing a conceptual design of a mobility system for use on the moon. This was to be done in conjunction with the planned objectives of conducting lunar exploration and associated functions with manned-unmanned capability and to be operational in the post 1980 period. The procedure followed in the initiation of the study program was the standard MSC request for a proposal to the summer design institute. The guidelines for the study were spelled out in the preliminary statement of work issued to the design institute fellows and was used to formulate a proposal for accomplishing the goals stated in the MSC request. In order to accomplish the proposed aims the summer study was divided into two distinct periods defined as Phase I and Phase II with specified tasks to be investigated in each phase. Figure 2-1 represents the path followed in the investigation during Phase I.

The institute was in progress for eleven weeks of which approxi-

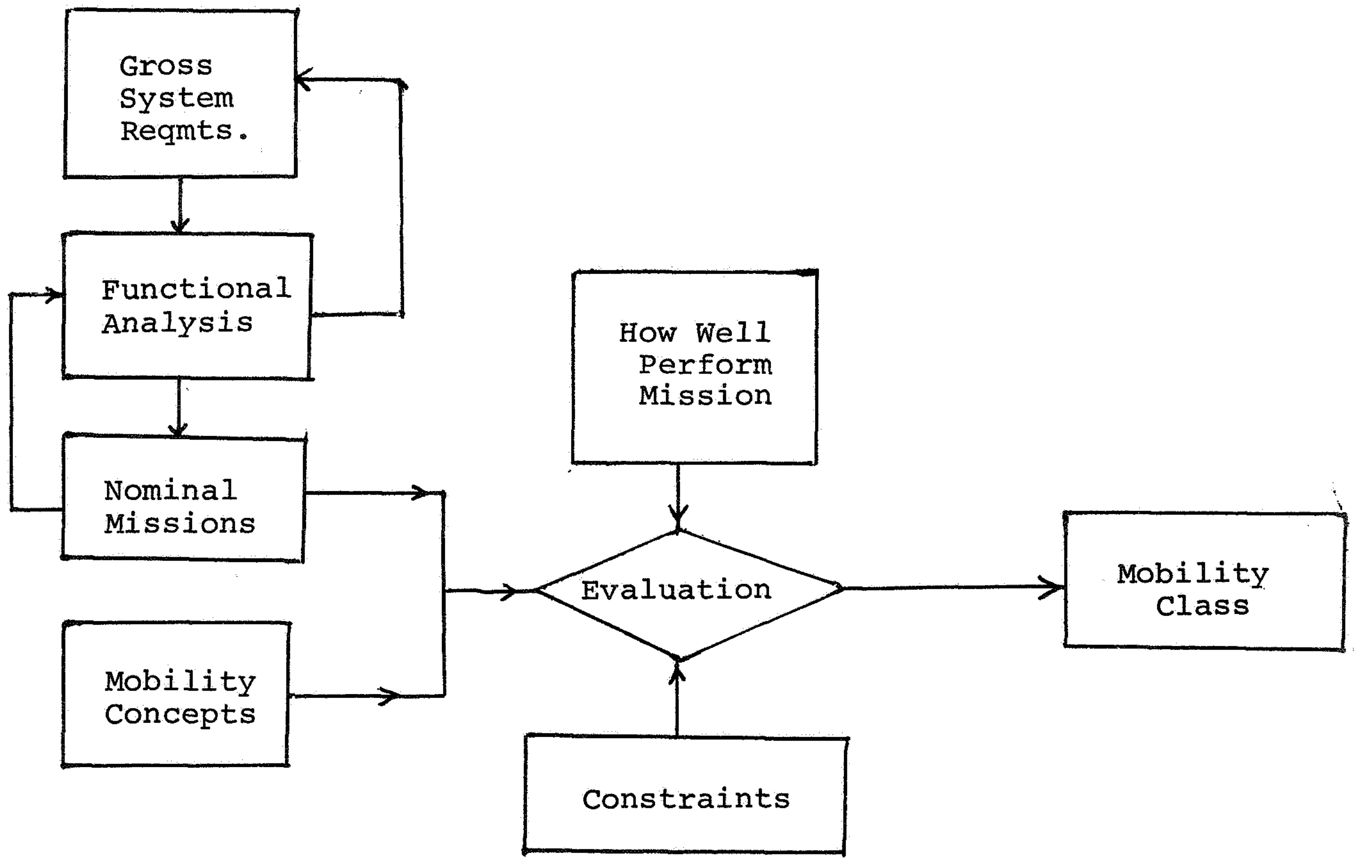


FIGURE 2-1. DEVELOPMENT OF PHASE I TASKS

mately forty-two days were scheduled for developing the problem, with thirty-eight percent of the time on Phase I and the remaining time on Phase II. The remainder of the allotted time was spent in familiarization with MSC facilities and personnel, systems engineering indoctrination, organization, problem background study, and analysis and interpretation of the work statement. The amended work statement is in Appendix A. This period encompassed the first two-and-one-half weeks of the summer study. It culminated with an informal presentation to MSC personnel directly concerned with the design problem during which the design team presented an amended version of the work statement and an overall method of an approach to solving the problem. The main goal of the Phase I period of study was to define and select a class of mobility system; and the goal for Phase II was the development of the selected concept.

2.1 Phase I

2.1.1 Objective of Phase I

The objective of Phase I of the summer institute was predicated on the NASA statement of work and its interpretation with modifications by the design team. The objective for completion by the end of Phase I was the definition and selection of a class of mobility system(s) for use in both manned and unmanned mode on the lunar surface and with potential for Mars use. The objective was broken down into five major tasks which were as follows:

(1) Define System Requirements.

(2) Functional Analysis to Fourth Level.

(3) Development of Candidate Systems.

(4) Establishment of Parametric Evaluation Criteria.

(5) Substantiation of Selection of Candidate System Class.

The above tasks to be developed during the Phase I time period represented 38 percent of the design team's time allotted for the summer study.

In order to accomplish the goals set forth in the statement of work by the given deadlines and to a degree of effort consistent with the work statement, an overall time schedule and level of effort plan were evolved. The documents, as prepared by the groups, were utilized to coordinate the team effort and each individual group's effort. The detailed overall task coordination time schedule is presented in Figure 2.1-1.

2.1.2 Development of Phase I Tasks

The design team's approach to attacking the problem was to subdivide into groups associated with the major design factors. This led to the formation of a configuration group, an astrionics group, and power and propulsion group and a human factors group.

Analysis of the Phase I tasks by each group soon revealed that several substudies were required in order to develop guidelines and data for completion of the tasks as stated in the NASA work statement. These substudies were generated by either the respective groups dealing with the problem or through the formation of committees which cut across group lines.

PHASE I: DEFINITION and SELECTION of MOBILITY SYSTEM

Task 1: Define System Requirements

Task 2: Functional Analysis of Candidate Systems

Task 3: Development of Candidate Systems

Task 4: Parametric Evaluation Criteria

Task 5: Substantiate Selection

PHASE II: DESIGN of SELECTED SYSTEM

Task 6: Concept Development

Task 7: Functional, Reliability Analysis

Task 8: Reports

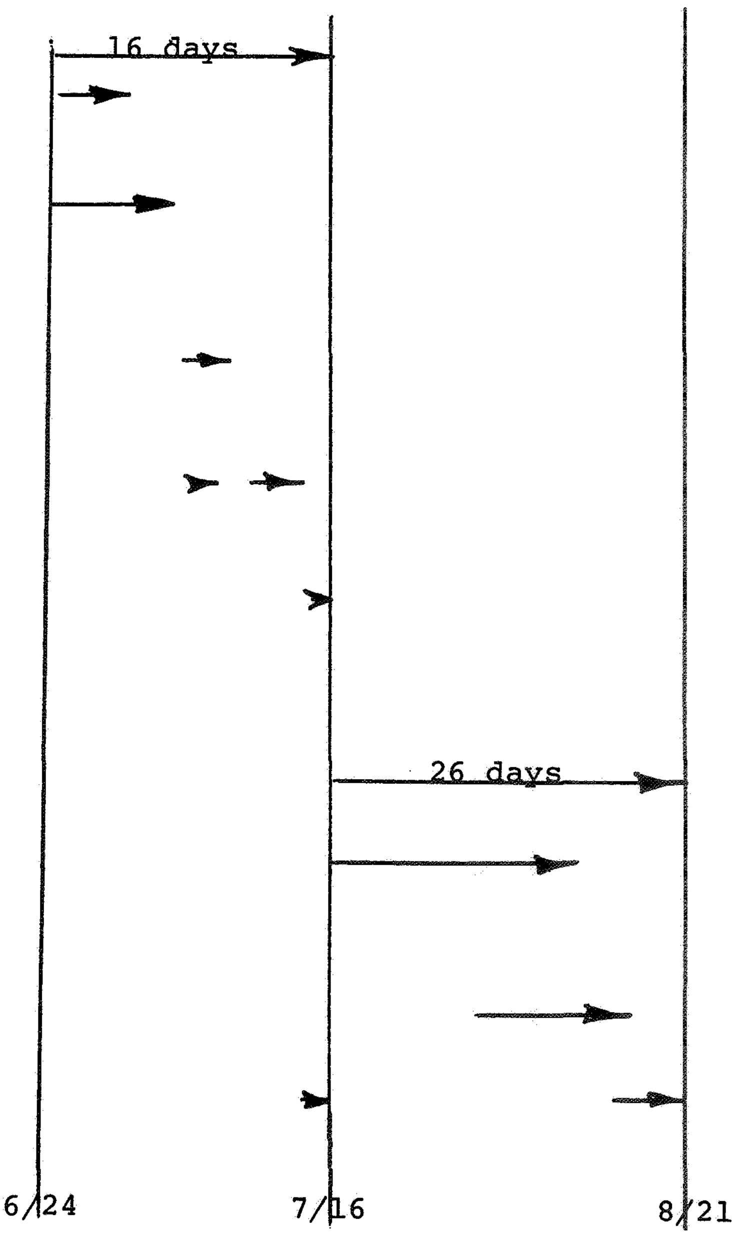


FIGURE 2.1-1 MASTER SCHEDULE FOR PHASE I-II SUMMER STUDY

Special studies were undertaken by members of the configuration and propulsion groups dealing with the gross mobility system requirements and lunar surface characteristics. Committees were created to prepare reports outlining and defining the parametric evaluation criteria and functional analysis for candidate systems. Parallel with the substudies undertaken, the respective groups defined the content of the tasks as they applied to their involvement in the study.

It was possible to derive a general functional analysis diagram for a mobility system to the third level; however, it soon became apparent that in order to define the gross system requirements, it was necessary to establish the mission of the mobility system on the lunar surface in the 1980-1990 period, consistent with the NASA-generated Integrated Program Plan. Accordingly, a more detailed study was undertaken which defined ten nominal missions that a mobility system would be required to perform during the time period in question. A summary of the missions developed is presented in Table 2.1-1.

Coincidental with the above studies the groups developed parametric studies in order to obtain data for comparison of mobility classes and determine the driving design factor(s). The parametric data generated included such items as the length of time an astronaut could perform EVA, cabin designs, power requirements, electronic sensing, propulsion requirements, communication requirements and systems, teleoperators, etc. These studies were generalized since, in essence, no specific mobility system was being considered but only subsystems and their requirements.

TABLE 2.1-1 SUMMARY OF MISSION DEFINITIONS

Mission Number and Mode	Description	Category
1 (Manned)	Drill Holes, Collect Core Samples, Deploy Science Stations (1670 lbs.)	Long Traverse
2 (Unmanned)	Continuous Collection and Transmission of Geodesy Data., Study of Lunar Fields, Mapping (500 lbs.)	Long Traverse
3 (Manned)	Same as Mission 2	Short Traverse
4 (Manned)	Deploy Drill or Science Station, Collect Rock Samples, Retrieve Drill and/or Core (1870 lbs.)	Short Traverse
5 (Manned)	Visual Exploration, Sample Collection, Photography (165 lbs.)	Local Sortie
6 (Manned)	Lunar Base Establishment, Construction, Grading	Lunar Base Support
7 (Unmanned)	Same as 6	Same as 6
8 (Manned)	Supply Lunar Base Support Maintenance, Fuel, Equipment	Same as 6
9 (Unmanned)	Same as 8	Same as 6
10 (Manned)	Transport Personnel Between Base and Tug	Same as 6

Table 2.1-2 and Figures 2.1-2 to 2.1-4 represent typical data generated during this time in the study.

2.1.3 Brainstorming

A method of conceptual design associated with systems engineering is a process commonly referred to as brainstorming. Brainstorming is an idea generating approach without any attempt to stifle creative thought as far as engineering constraints are concerned. The results of this creative thinking process yielded nineteen initial concepts which are listed in Table 2.1-3. However, examination of these concepts indicated that they fell within four main classes, namely, Rover, GEM, Flyer and Hopper. Chapter 7 describes these mobility systems. Based on this classification the groups continued along a generalized mobility class conceptual approach rather than a specific mobility design.

TABLE 2.1-3. LIST OF BRAINSTORMING CONCEPTS

Rover	Ballistic Package (Cannon)
Crawler	Hopper
Gyro Stabalized Bicycle	Bouncing Ball
Screw Driven Platform	Flyer-GEM Hybrid
Snowmobile	Rover-Flyer Hybrid
Rolling Ball	Rover-GEM Hybrid
Rolling Cylinder	Rimless Wheel
Mechanical Horse	Self Generated Fiber-Glass Tube
Dust Jet	GEM
Flyer	

TABLE 2.1-2 ELECTRICAL POWER REQUIREMENTS FOR CANDIDATE SYSTEMS

Mobility System	Astrionics	Science	Motion	Environment	Total
Rover	0.5	0.5*	2.0	0.5**	3.5KW
GEM	0.5	0.5*	0.1	0.5**	1.6KW
Flyer	0.6	0.5*	0.1	0.5**	1.7KW

* Value does not include drilling or construction.

** Unmanned changes value from 0.5 to approximately 0.1 KW on all candidates.

Drilling requirements: 3KW

Construction requirements: 1KW

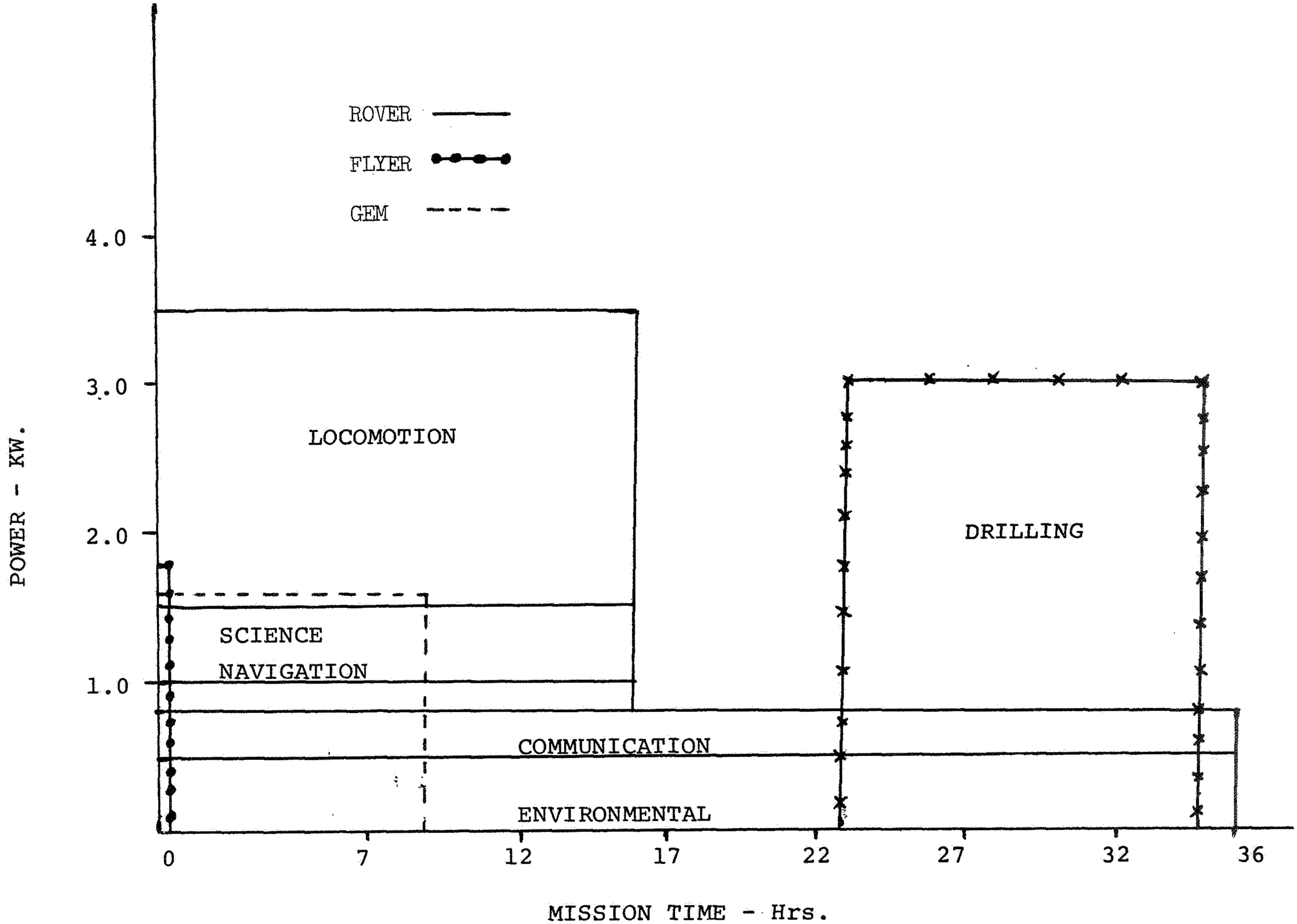


FIGURE 2.1-2. ENERGY REQUIREMENTS FOR 36 HOUR MANNED MISSION

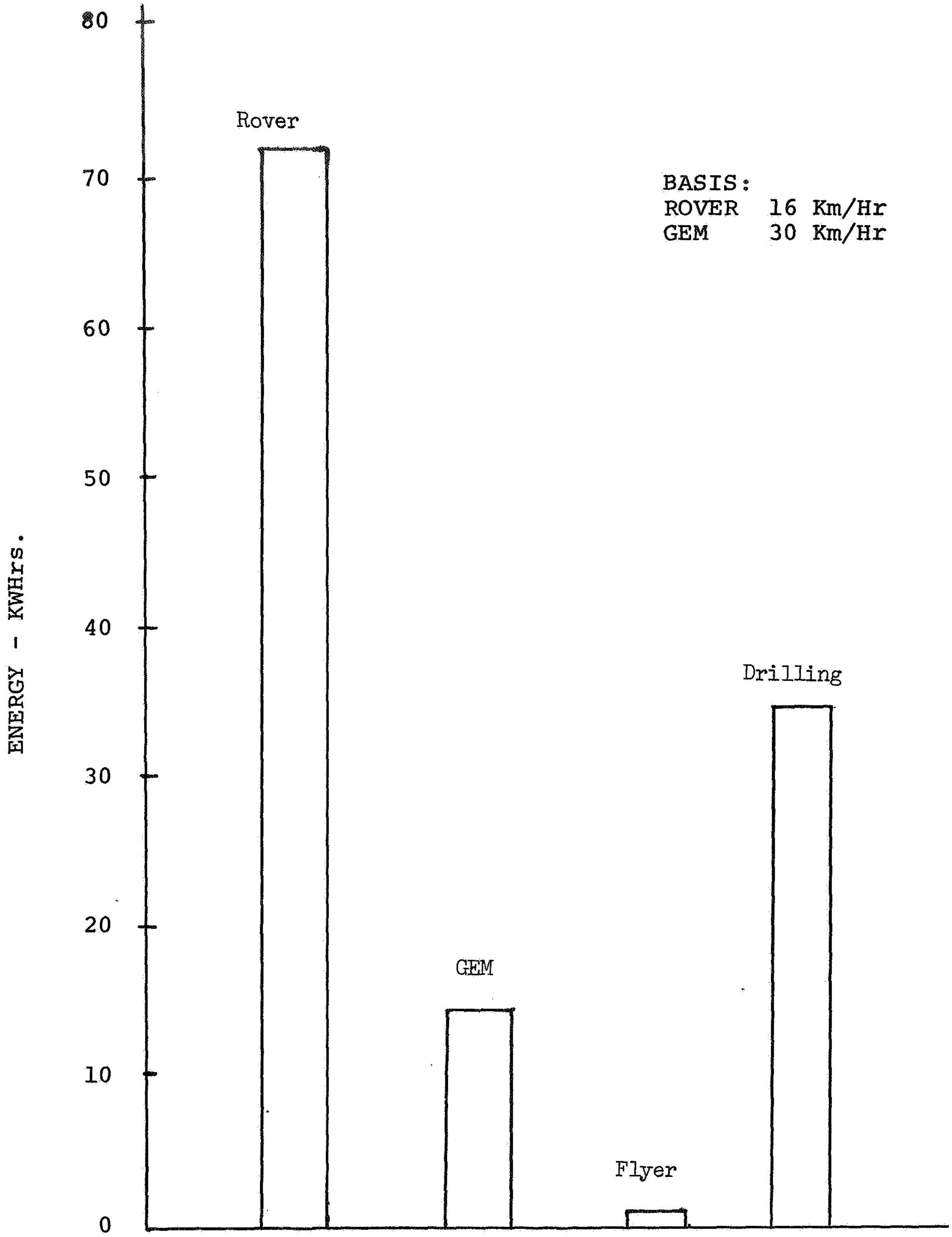


FIGURE 2.1-3. ENERGY REQUIREMENTS FOR 250 Km. MANNED MISSION

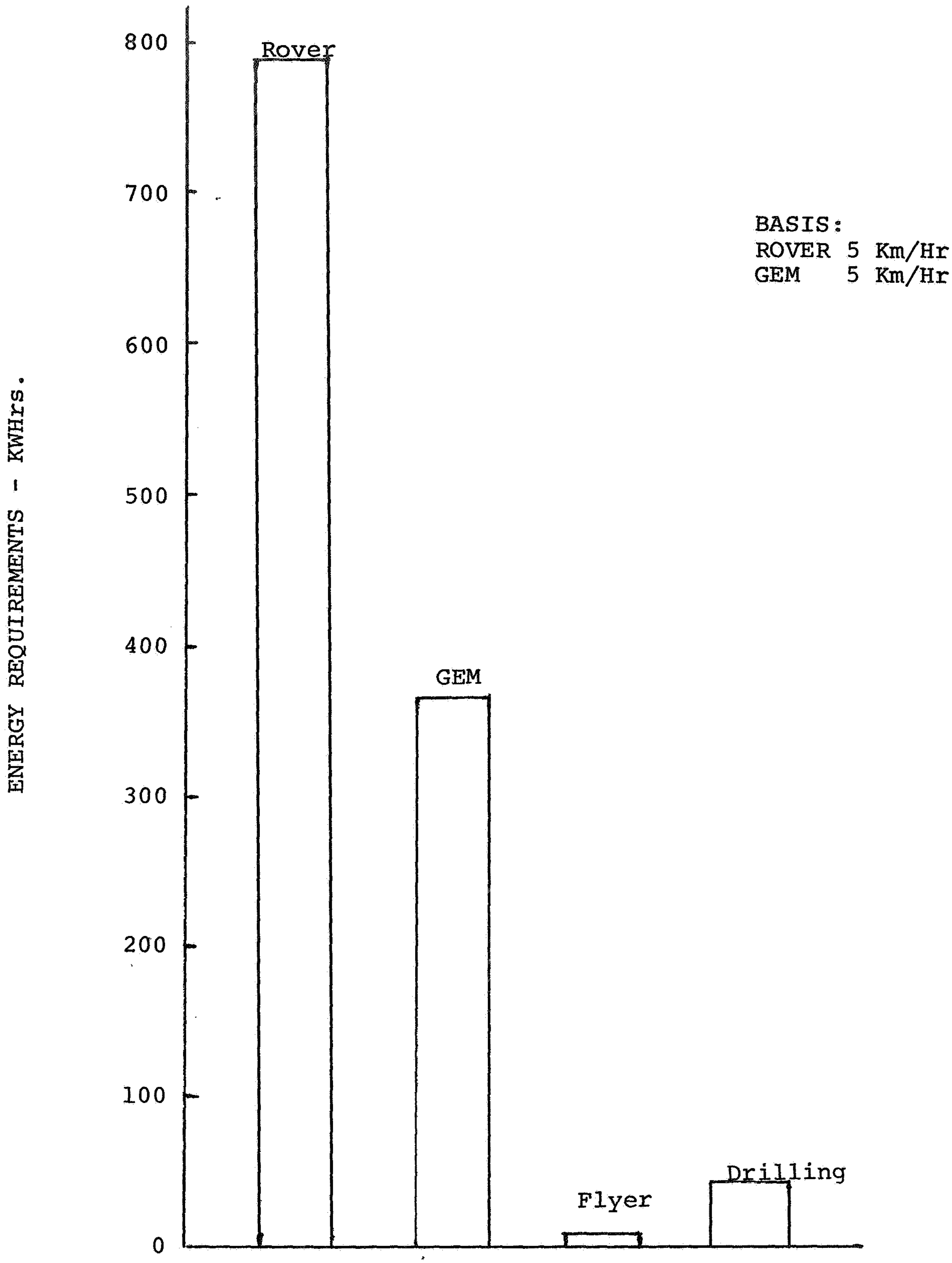


FIGURE 2.1-4. ENERGY REQUIREMENTS FOR 1500 Km. UNMANNED MISSION

2.1.4 Criteria for Evaluation

A dual method of approach was proposed and developed to establish criteria for evaluating the various candidate mobility systems. The methods which were utilized were the Evalumatrix technique, subjective in nature, and the Cost-Effectiveness approach which is an objective approach.

The Evalumatrix technique used a matrix in which the parameters deemed most important for the mobility system design were developed and weighted by the respective groups. Selection of the recommended mobility system classes for continued study in Phase II was based on this method. Figures 2.1-5 and 2.1-6 are examples of the Evalumatrix method used in the study.

The Cost-Effectiveness technique was designed with the aim of developing a mathematical model from which cost-effectiveness data could be generated for any candidate mobility system for comparison purposes. A completely rigorous, tested model was not completed by the end of Phase I, but preliminary studies indicated this approach could be a powerful analysis tool.

2.1.5 Phase I Results and Recommendations

Phase I was concluded with a formal presentation to update NASA-MSC personnel involved with the institute design problem. The presentation consisted of an oral portion during which a survey of the work performed was presented in brief, and the formalized documented written study was distributed.

The results of the study are as follows:

Performance Parameters	Weight Factor	Mission Type	Mission Weight	Mobility Class			
				Rover	GEM	Flyer	Hop
Payload / Gross Wt.	40	1	0.20				
		2	0.45				
		3	0.20				
		4	0.15				
Fuel (Non-Stop) / Fuel (Str.-Stp.)	20						
Kilometers / Lb. of Fuel	20						
Speed ratio	20						
	Sum			71	58	32	32
	Normalized			100	82	45	45

FIGURE 2.1-5. POWER AND PROPULSION EVALUMATRIX

Evaluating Group	Candidate Mobility System Class			
	Rover	GEM	Flyer	Hopper
Power and Propulsion	100	82	45	45
Astrionics	100	98	78	58
Human Factors	100	83	72	42
Configuration	100	76	60	43
Average	100	85	66	47

FIGURE 2.1-6. EVALUMATRIX RESULTS OF MOBILITY CLASSES

(1) The flyer class mobility system was eliminated due to excessive propellant requirements for the constraints imposed.

(2) The hopper class mobility system was eliminated due to crew safety considerations, control reaction times, and payload limitations.

(3) Manned-Unmanned mobility systems require a moon satellite communications system.

(4) The Evalumatrix results as well as the Cost-Effectiveness preliminary study indicates a Rover class vehicle is the most adequate with the GEM class a close second.

(5) Based on the Mission Requirement Study and the Integrated Program Plan fifteen vehicles are required for lunar exploration during the 1980-1990 period.

Consistent with the above results the following statements presented to NASA-MSC represented the recommendations of the design team for the direction of the study in Phase II:

(1) Develop a "Near Ground System" which is to be interpreted as a GEM or Rover class mobility system.

(2) Emphasis on GEM class mobility system.

(3) A continuance of the study on Cost-Effectiveness.

2.2 Phase II

The Phase II portion of the study was based on the recommendations put forth by the design team at the conclusion of Phase I. In light of the tasks to be accomplished it was decided to reorganize the groups. The human factors group was disbanded and its members incorporated into the remaining three groups. The philosophy behind the reorganization was the idea of forming systems teams within the individual groups in order to approach the design problem from an overall view, expand the capability of the groups and maximize interaction.

A systematic analysis of the final aim of Phase II led to the identification of definite areas which required in depth study to develop and specify a final design of the mobility system . Based on the analysis, a method of approach was evolved and followed as illustrated in Figures 2.2-1 and 2.2-2. Table 2.2-1 gives the specific charge given to each of the committees concerned with the design development.

Phase II was concluded with an oral presentation to MSC personnel outlining the conclusions and recommendations of the summer study. In addition the written documented material presented in this report was edited and finalized.

2.2.1 Synthesis of Candidate Systems

The intent behind the formation of the synthesis committee was an attempt to develop a method of approach for the logical basis of design of the final mobility system from a large number

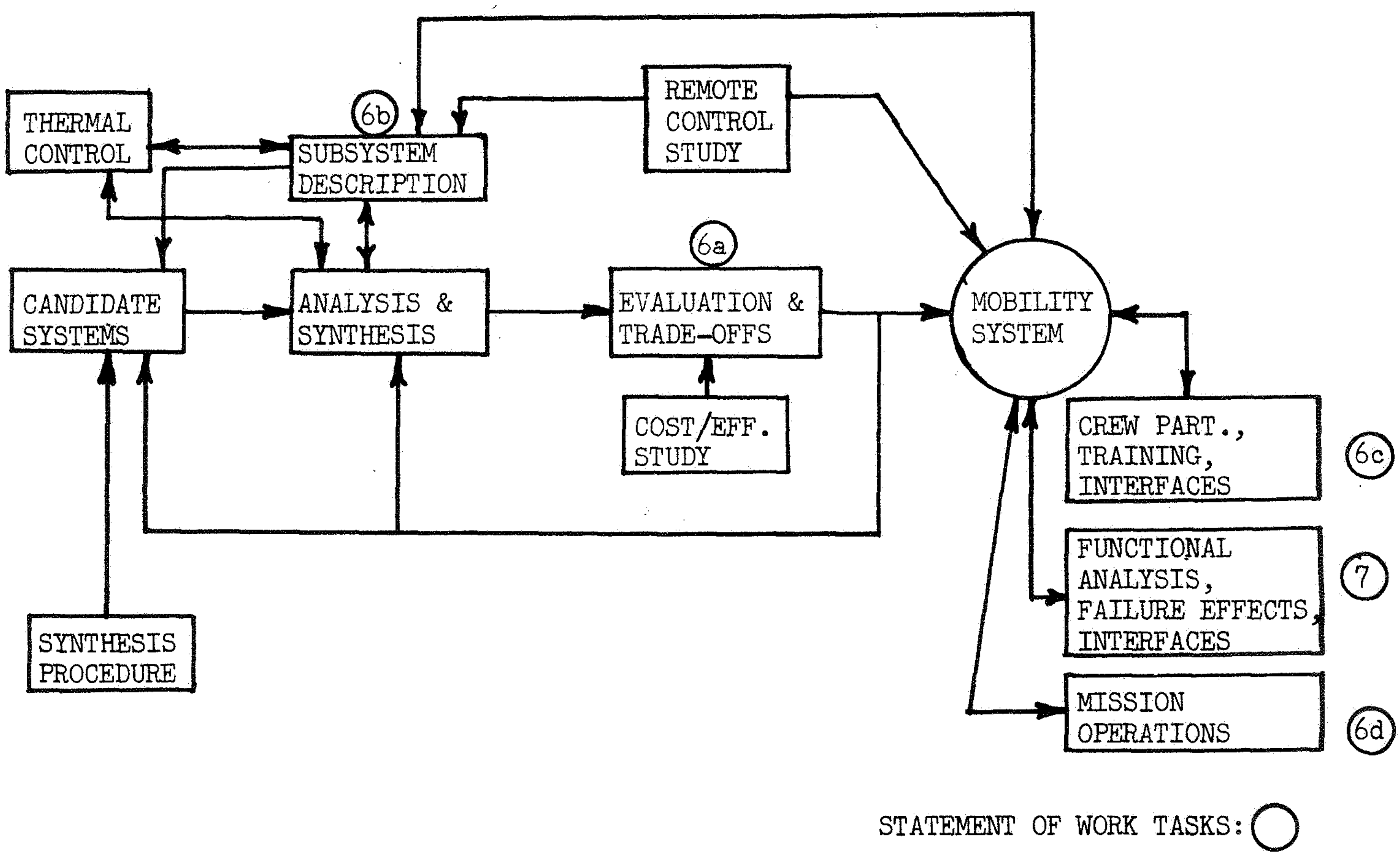


FIGURE 2.2-1 PHASE II TASK DEVELOPMENT

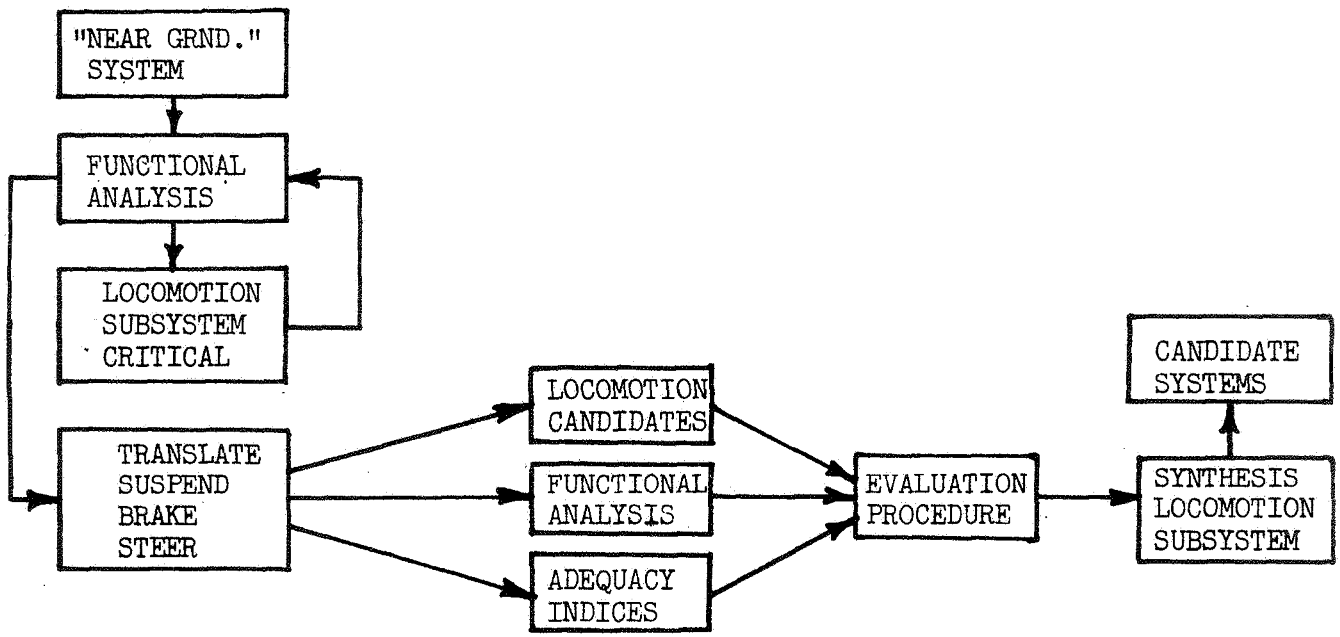


FIGURE 2.2-2. DETAILED SYNTHESIS PROCEDURE EVOLUTION

TABLE 2.2-1. LIST OF COMMITTEES AND CHARGES

Committee	Charge
Synthesis of Candidate Systems	Establish procedure for synthesizing candidate mobility systems.
Subsystem Description	Produce detailed description of subsystem by function, input, output, location, constraints relative to other subsystems and physical environment and interactions.
Remote Control Problems	Investigate remote control problems, lunar visibility. Establish trade-off possibilities, indicate problem areas and alternatives.
Evaluation Effectiveness	Improve cost-effectiveness technique, prepare for use on candidate systems, generate sample results.

of candidates. The technique was predicated on first establishing the procedures by which one would synthesize the candidate mobility systems from subsystems, synthesis of the mobility systems, evaluation of the candidates by parametric studies to reduce the number, and selection of a final mobility design.

As the study evolved it became apparent that the driving design factor in the synthesis of any mobility system was the locomotion mode. The synthesis technique developed was modified from an overall mobility systems view to one dealing with a process for synthesizing locomotion subsystems with the goal of designing the mobility system around the locomotion subsystem.

In order to insure unbiased evaluation of candidate locomotion modes, an evalumatrix method was developed. This method was based on Adequacy Indicies which were associated with the important functions and requirements of the locomotion systems. Such items considered were the statement of work, gross systems requirements, functional analysis and design group guidelines. Tables 2.2-2 and 2.2-3 list the Adequacy Indicies and Locomotion Candidate Evaluations. The final recommended locomotion candidates for more detailed analysis were a half track, plenum tracks and full tracks.

2.2.2 Subsystem Description

In order to insure the completeness of the mobility system an identification procedure was developed which described and defined all the subsystems to be included. The procedure followed was to develop the subsystems from an initial black box concept with a specification of the input-outputs and subsystem interactions. The driving design factors for the black box subsystems were developed and analyzed by development of parametric studies until the black box was completely described as to its capability to perform its required function.

Based on the requirements description, trade off studies were undertaken to specify the appropriate subsystems from what was available in existing programs or among subsystems designed by the team groups. This procedure led to the recommendation of the best existing hardware to perform a given task of a generalized design for the development of a specific subsystem or component.

TABLE 2.2-2. ADEQUACY INDICIES AND WEIGHTING FACTORS

	Adequacy Indicies	Weight
I.	Performance While Suspending	20 Units
II.	Performance While Translating	20 Units
III.	Performance While Steering	20 Units
IV.	Performance While Braking	20 Units
V.	Active Provision of Translation	10 Units
VI.	Active Provision of Steering	10 Units
VII.	Active Provision of Braking	10 Units

2.2.3 Remote Control

The remote control studies were concerned with such diverse areas as the role of manipulators and teleoperators, lunar visibility problems, obstacle avoidance, navigation, time delays, predictor display operations, and associated problems with manned - unmanned

TABLE 2.2-3. RESULTS OF LOCOMOTION CANDIDATE EVALUATION

Candidate	Total Units	Effectiveness
	Single Systems	
Track	106	0.883
Wheel	96	0.800
Auger (Screw)	82	0.683
Shaft	72	0.600
Jointed Leg	72	0.600
Ski	70	0.583
Ball + Socket	54	0.450
Plenum	53	0.441
Reaction Jet	30	0.250
	Combination Systems	
Plenum + Tracks	111	0.925
Plenum + Wheels	100	0.833
Wheels + Tracks	107	0.892
Plenum + Reaction Jet	83	0.692
Plenum + Jet + Tracks	111	0.925

moving modes. The main consideration in the analysis of the various areas was: Which could perform a function best - a man-machine combination, man, or machine? The solution to many of the problems was combinations of all modes of operations.

Among the important factors incorporated in the final report are: a laser-based sensing device is recommended for obstacle avoidance; time delays govern unmanned operations so that a Lunar Orbiting Space Station control is required; predictor display control of the vehicle is required for maximizing speed; two TV cameras are required; head light illumination is necessary; and a combination Halo-Hummingbird Libration Point Satellite System should be in operation for full lunar coverage.

2.2.4 Cost Effectiveness

The model developed in Phase I was refined and tested. Parametric sensitivity was investigated and the model was tested on known programs with good agreement. The model was used to give cost-effectiveness data on the recommended mobility system. The analysis indicates a total program cost of 3.9 billion dollars with a first unit cost of 27 million and an effectiveness of 0.98. The total program cost is for fifteen vehicles over a ten years period and includes research, development, testing and evaluation, production, consumables, and man operating-time costs.

2.2.5 Team Group Activities

At the conclusion of the committee studies and based on the recommendations of the reports issued, the three locomotion candidates were expanded into individual completely designed mobility units. This procedure necessitated the disbanding of the committees and having their respective members return to their designated groups.

Power, Propulsion and Thermal Control: This group concerned itself with the locomotion requirements, thermal constraints, life support systems, cabin design and the power requirements as related to the three candidate systems. Trade-off studies were undertaken in each area and parametric evaluations were generated. Based on mission requirements the power and locomotion demands were calculated for both the manned and unmanned mission. The governing design conditions are for the unmanned case since it places the greatest demand on the mobility system. Table 2.2-4 lists the specified parameters for the finalized mobility system requirements.

TABLE 2.2-4. SUMMARY OF POWER AND LOCOMOTION REQUIREMENTS

Parameter	Manned	Unmanned
Peak Power (KW)	5.0	3.9
Average Power (KW)	4.2	2.2
Average Speed (Km/hr)	10	5
Total Propellant Required (lbs) * (Oxygen-Hydrogen)	112	925
Soil Slope Design (Degrees)	3	3
Mission Time (Hrs)	36	420
Total Mission Energy Requirements (KWH)	112	924

* Includes 25% above calculated value (Safety Factor).

Astrionics: The astrionics group investigated such subsystems as communications, TV requirements, vehicle control, navigation and guidance, obstacle avoidance, computer requirements, and communication and control satellites. Parametric studies and hardware investigations led to specifications of the subsystems dealing with the astrionics areas.

Configuration: The configuration group designed and analyzed the structure, integrated the subsystems, and produced the drawings for each candidate mobility system. The integration of group inputs included such items as weight analysis, cabin configuration and content, astrionics package and location, locomotion and power requirements, and environmental requirements and constraints of subsystems. The three conceptual designs were completed and, based on the groups' evaluation criteria, were reduced to only one final mobility system. The halftrack, and plenum - tracks were eliminated. The plenum - tracks concept proved too costly from a propulsion point of view due to inability to minimize the plenum gas losses. The halftrack was undesirable from the locomotion penalty incurred due to the combination drive system. Since both wheels and tracks are present, one must design independent locomotion systems for each mechanism. In essence the dual mode (hybrid) system incurs locomotion penalties since one can not readily integrate the drive mechanism for locomotion but must essentially design two independent units for the same mobility vehicle. The recommended vehicle is the fully tracked mobility system. Details on dimensions and weight are presented in Table 2.2-5 and the conceptual drawing in Figure 2.2-3.

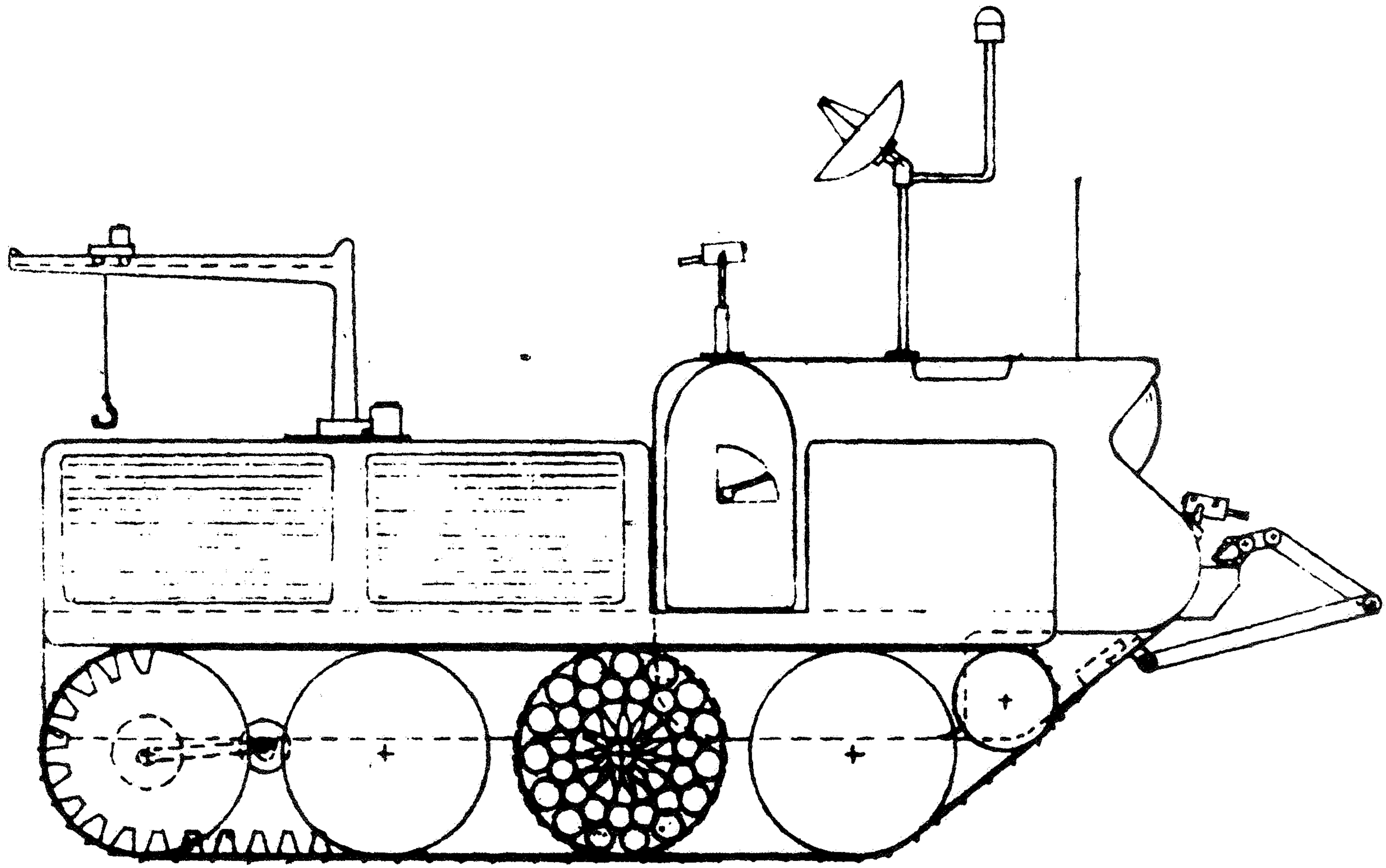


FIGURE 2.2-3 CONCEPTUAL DRAWING OF LUNAR MOBILITY SYSTEM (MULE)

TABLE 2.2-5. DIMENSIONS AND WEIGHTS OF MULE

Overall Width	10.5 Ft.
Overall Length	21.8 Ft.
Overall Height	9.5 Ft.
Track Width	2.0 Ft.
*Track Length	13.7 Ft.
Track Height	4.0 Ft.
**Dry Weight	5985 Lbs.
**Gross Weight	9705 Lbs.

* Ground contact length.

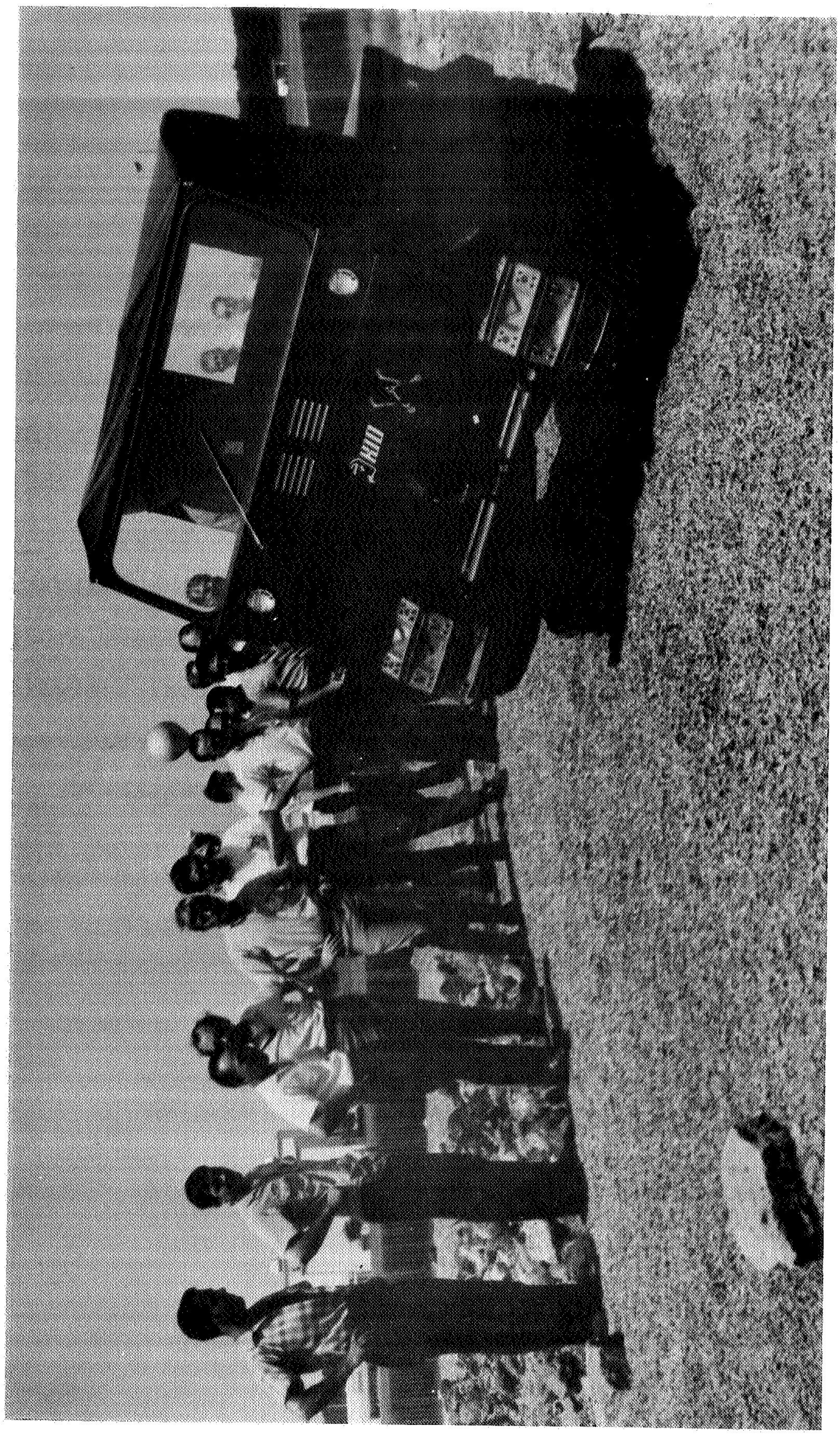
** Weights are based on unmanned mission requirements and gross weight includes crew weights.

It is interesting to note that a vehicle employing a locomotion mode similar to the one recommended in this study is available. The vehicle is called the KID and is manufactured by the Kinematics International Division of LTV. A demonstration of the KID was arranged for and conducted at the simulated lunar surface facilities on the MSC site. The test indicated the feasibility of the locomotion system for a lunar mobility system.

PART I

REQUIREMENTS

To design is to formulate a plan for satisfaction of requirements; thus, engineering systems design begins by specifying the requirements to be placed on the system. Initially gross system requirements are specified and during the design process the requirements are specified in more detail. Chapters 3 through 6 present the requirements for MULE and discuss the processes by which these requirements were defined. Chapter 3 presents the "Statement of Work" provided by MSC. It was obvious that the environment of the lunar surface and the tasks to be performed on the lunar surface would be critical factors in the design; thus, studies were made early to determine these. These studies are presented in Chapters 4 and 5. Formal functional analysis was used throughout the system design to specify the requirements in more detail and this analysis is discussed in Chapter 6.



CHAPTER 3

WORK STATEMENT AND PROPOSAL

J. E. Robertshaw

3.1 Work Statement

The statement of work (SOW) was presented to the Institute by the Manned Spacecraft Center (MSC). The Institute was instructed to consider the SOW as a request for a proposal for work. The original form of the SOW is presented in Appendix A. During the preparation of the proposal for work and during the study itself, changes and clarifications were made. They are indicated by parentheses with an asterisk ()*. The changes and clarifications are listed immediately following the SOW.

3.2 Proposal

3.2.1 Introduction

A proposal for work to be done by the Institute was presented to MSC in a two-hour oral presentation on June 23. The proposal was based on the SOW as amended and clarified during negotiations between the Institute and MSC.

The content of the proposal may be divided into two parts. The first part consisted of objectives and constraints required by the SOW. The second part consisted of the philosophy and procedures which the Institute would use to attack the problem. The content of these two parts is summarized in the following sections.

3.2.2 Summary of SOW Requirements

The primary objectives of the study were to design a mobility system for manned and unmanned operations in the post 1980's and to establish research and development requirements for the proposed system. The system was to be used for lunar exploration, for science experiments, for the support of lunar base construction, and to develop mobility concept capability for planetary operations. The prime mode of delivery of the system was to be the Space Tug (see Chapter 5), and it was assumed that a lunar space station would be in lunar orbit.

The mobility system was to have the following performance capabilities.

o Negotiate	30° slopes
	50 cm obstacles
	50 cm steps
	90 cm crevices
o Pitch and roll stability up to 45°	
o Range	250 km manned
	1000-1500 km unmanned
o Payload	400 lb/man for EVA
	200 lb/man for shirtsleeve environment operations
plus	1000-2000 lb science equipment

In addition to the above performance requirements, the system was to have the following operational capabilities.

- o Operate in lunar and Mars night
- o Lunar operations in post 1980's
- o Lifetime 1 year
- o Operate in manned and unmanned modes
- o Manned mode - 36 hr plus 12 hr contingency
- o Crew could survive one failure in a subsystem

The manned mission mode was to be considered primary and planetary operations were to be considered from the standpoint of growth potential.

The primary design constraints were to be the following.

- o Dry weight less than 5000 lb
- o Ground clearance of 50 cm
- o Crew size of two
- o Compatible with Space Tug

3.2.3 Philosophy and Procedures

The Institute efforts were to be directed toward a concept definition study rather than a preliminary design study. Accordingly, the Institute was to emphasize methodology development over detailed specification of the systems and the configuration. The Institute was to channel a major portion of its effort into the development of system evaluation and system synthesis procedures.

The primary methodology to be used by the Institute was formal systems analysis. Functional analysis would be used to determine functional requirements, and trade-off techniques would be used to optimize system and subsystem capability. Emphasis was to be placed on simplicity and redundancy at all levels of the system, and consideration was to be given to potenital commonality of equipment for various missions. All classes of mobility systems were to be considered.

The Institute was to organize itself into groups. Each group would have a group leader directly responsible to the project director, who in turn was directly responsible to the project managers. The proposed organization is summarized in Figure 3.2-1. (During the second phase of the study, an organization deemed more consistent with the work to be done in phase II was adopted. This second organization is summarized in Figure 3.2-2).

The proposed management responsibilities and authority within the organization are summarized below.

1. The <u>study managers</u> shall be jointly responsible for the overall study directions and for the production of a technically meaningful report. They shall have authority as required to implement the above.
2. The <u>study director</u> will be responsible for the conduction of the study and the production of the final report. He shall have the authority to direct the Systems Design Institute personnel to implement the above. His direction will be subject to review by the study managers.
3. The <u>subsystem managers</u> (<u>project aides</u>) will be responsible for their assigned subsystem (group) and will report directly to the study managers. They shall have limited authority to give directions to the group leaders. These

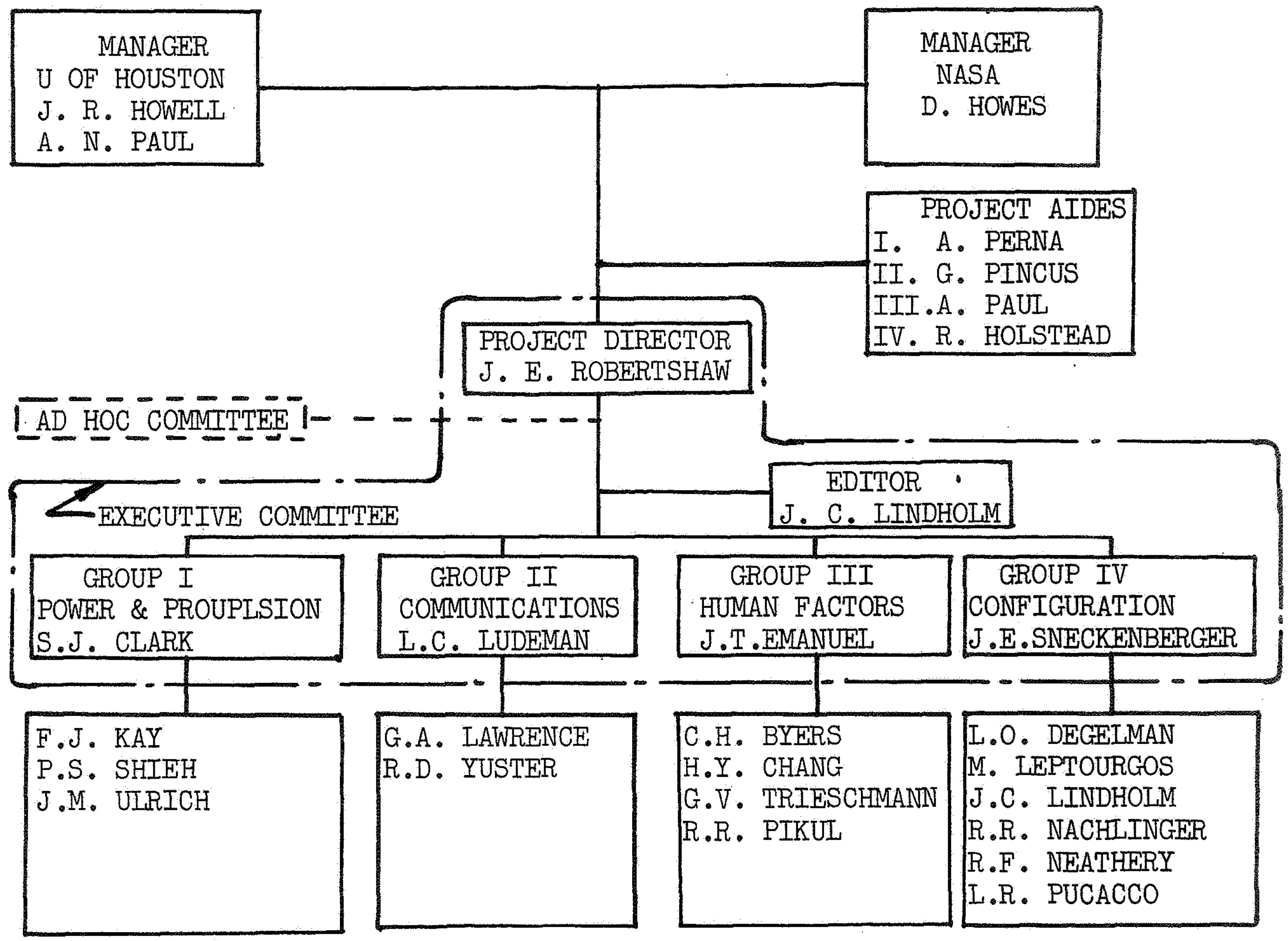


FIGURE 3.2-1 PHASE I ORGANIZATIONAL CHART

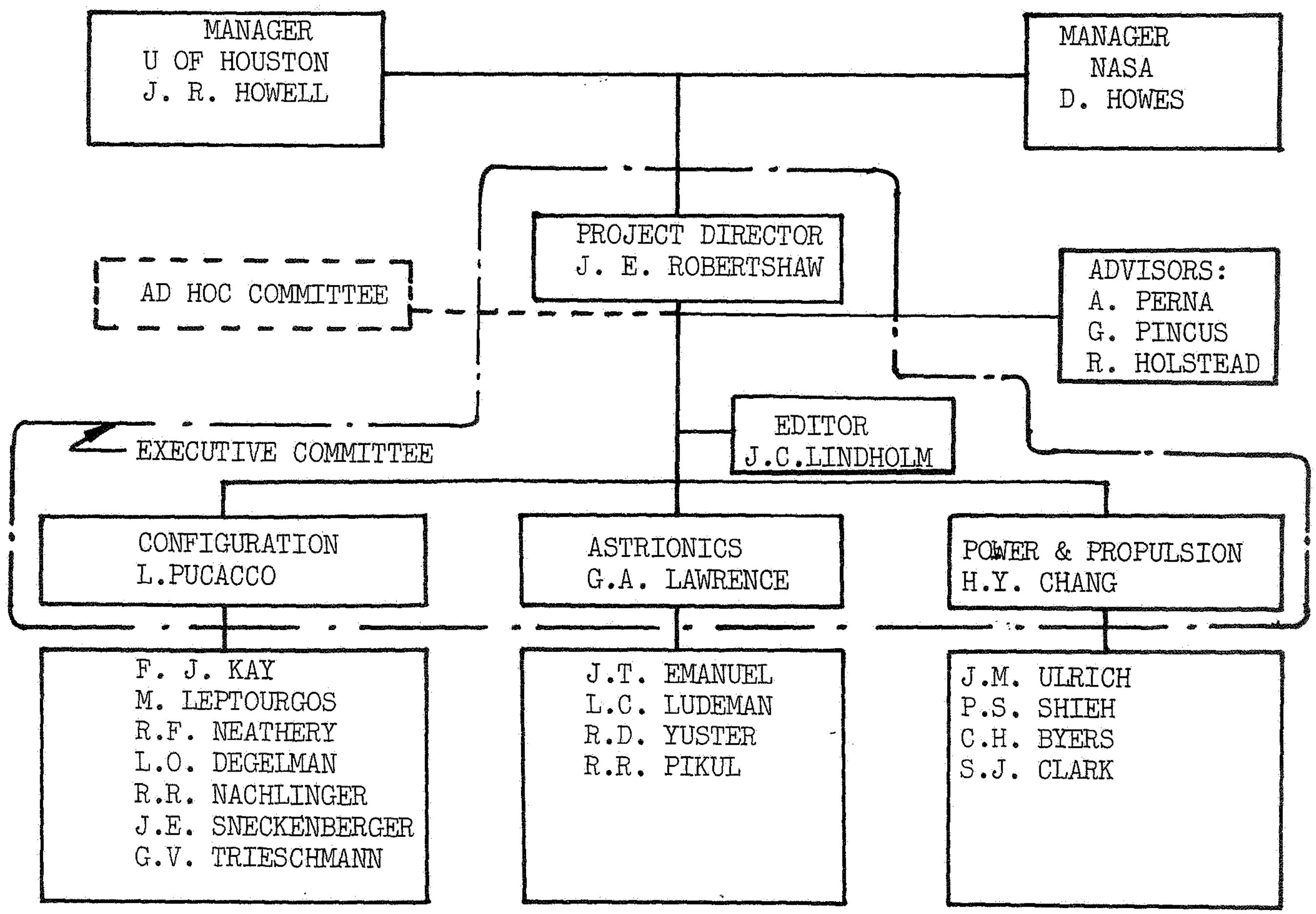


FIGURE 3.2-2 PHASE II ORGANIZATIONAL CHART

directions will be subject to review by the study managers and director. (In the second phase of the study the project aides were placed under the project director so that they reported directly to him).

4. The group leaders shall have the responsibility and authority for directing the activities of the Systems Design Institute personnel within their group.

The Institute proposed a breakdown of tasks in agreement with those in the SOW. A task coordination schedule was proposed which was subsequently changed slightly during discussion with MSC after the proposal presentation. A summary of the tasks is given below, and the final schedule and a work flow diagram are given in Figure 3.2-3 and Figure 3.2-4, respectively.

TABLE 3.2-1 TASK SUMMARY

Phase I: Define system requirements and select a class of mobility system.

Task
1. Define gross system requirements
2. Perform functional analysis
3. Define candidate systems
4. Establish evaluation criteria
5. Substantiate selection of the system

Phase II: Develop the system through configuration and subsystem trade-offs and analysis.

Task
6. Develop mobility concept
    a. Illustrate physical configuration with

sketches and drawings and perform design trade-offs.

b. Define the subsystems

c. Define crew participation, interfaces, and training.

d. Perform mission operations analysis

7. Expand functional analysis and requirements, document interfaces, perform failure and effects analyses.

8. Prepare oral and written papers.

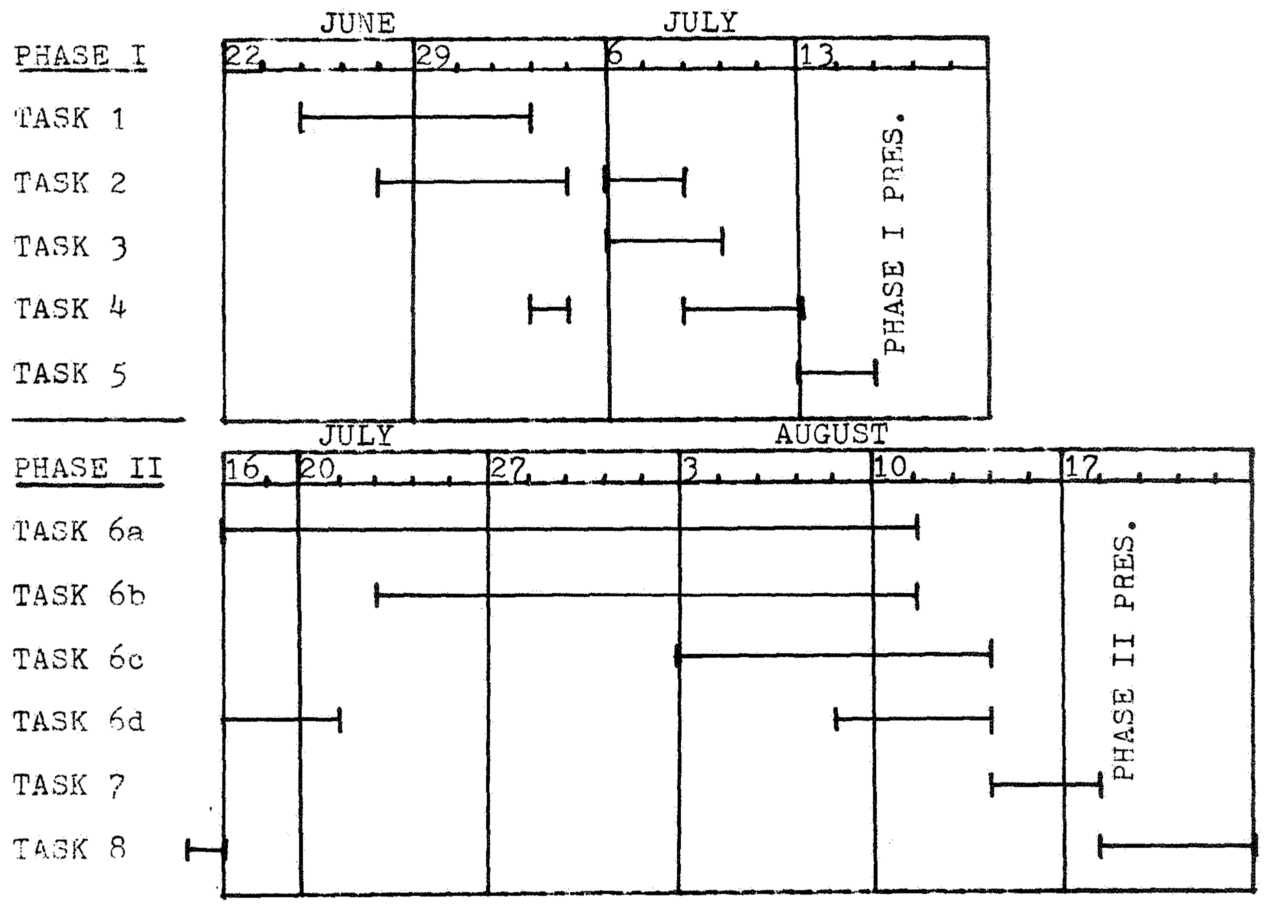


FIGURE 3.2-3 TASK COORDINATION SCHEDULE

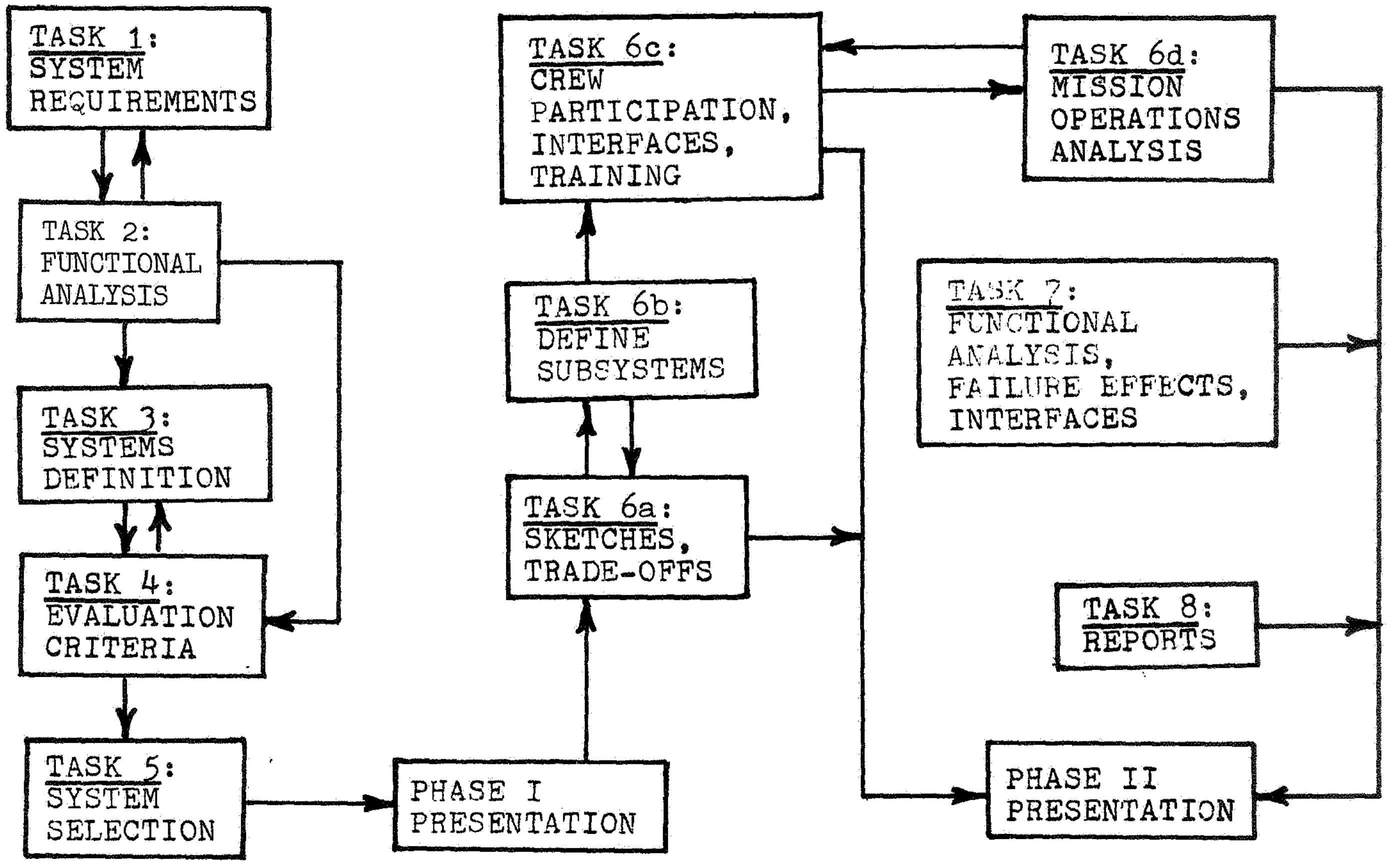


FIGURE 3.2-4 WORK FLOW DIAGRAM

CHAPTER 4

LUNAR AND MARS ENVIRONMENT AND TERRAIN

S. J. Clark

4.1 Introduction

All design solutions must satisfy two basic requirements: (1) the device must adequately perform the functions for which it was designed and (2) the device must continue to perform when exposed to its application environment. These requirements become very critical in the design of components and assemblies for space.[1]

The environmental and terrain factors that are most important in the design of a space surface mobility vehicle for the moon are:

1. Reduced gravitational effects.
2. Lack of ambient pressure.
3. Temperature extremes.
4. A rough terrain covered by blocks, craters, and various types of ridges and valleys.
5. Visibility problems.

Table 4.1-1 was assembled[2,3] to permit rapid comparisons between some of the more important physical constants for earth, moon, and mars.

4.2 Reduced Weight

From Table 4.1-1 it can be noted that the gravitational acceleration on the moon and mars is 5.31 and 12.3 feet/sec^2 respectively. This causes some problems with respect to vehicle design for these environments. The main problem evolves from the fact that surface bound vehicles obtain their translation, braking, and steering ability from surface contact forces. This presents some problems since inertial forces are constant while the weight forces are reduced to about 1/6 on the moon and less than 1/3 on Mars.

For all practical purposes the moon has no atmosphere (10^{-10} to 10^{-13} of that of the earth) Since it has no atmosphere, it is continually bombarded by micro-meteorites, ultraviolet radiation and solar corpuscular and primary cosmic rays.[4]

The general approach for designing for a vacuum is as follows.[1]

1. Minimize the contact of moving surfaces (use flexure pivots where possible)
2. High vapor pressure metals such as cadmium, zinc, and selenium are protected by coatings of low vapor pressure materials such as silver or gold.
3. Sensitive optical surfaces are located to minimize their exposure to redeposition of evaporating fluids of materials.

4.3 Thermal Environment

From Table 4.1-1, one can note that the surface temperature extremes are tremendous on the lunar surface (-279 to +243 ^{o}F) Due to lack of an atmosphere, heating and cooling design must rely entirely on conductivity and radiation for heat exchange.

TABLE 4.1-1 PHYSICAL CONSTANTS FOR EARTH, MOON AND MARS

Parameter	Earth	Moon	Mars
Diameter, Nautical Miles	6888	1876	3680
Diameter, Kilometers	12756	3476	6820
Equatorial Diameter Ratio	1.0	0.2723	0.535
Density, lb/ft^3	344.6	207.5	243.0
Density, gm/cm^3	5.517	3.33	3.89
Mean Density Ratio	1.0	0.6043	0.705
Mass, lbm (10^{23})	131.79	1.621	14.22
Mass, gm (10^{26})	59.77	0.735	6.45
Mass Ratio	1.0	0.01226	0.1069
Gravity, ft/sec^2	32.19	5.31	12.3
Mean Surface Gravity Ratio	1.0	0.165	0.39
Gravitational Parameter ($GM = \mu = g_o R^2$)			
ft^3/sec^2 (10^{15})	14.08	0.173	1.516
km^3/sec^2 (10^4)	39.86	0.490	4.293
Albedo	0.36	0.07min.	0.15
Surface Temperature Range (min/max)			
^oF	-90/140	-279/243	-193/90
^oC	-62/60	-173/117	-125/32
Atmospheric Pressure at Surface, psi	14.7	nil	0.8 - 2.0
Atmospheric Constituents	78% N_2 21% O_2 1% A	none	90% CO_2
Miscellaneous Information			Clouds of haze and dust, storms with wind velocities to 60 mph.

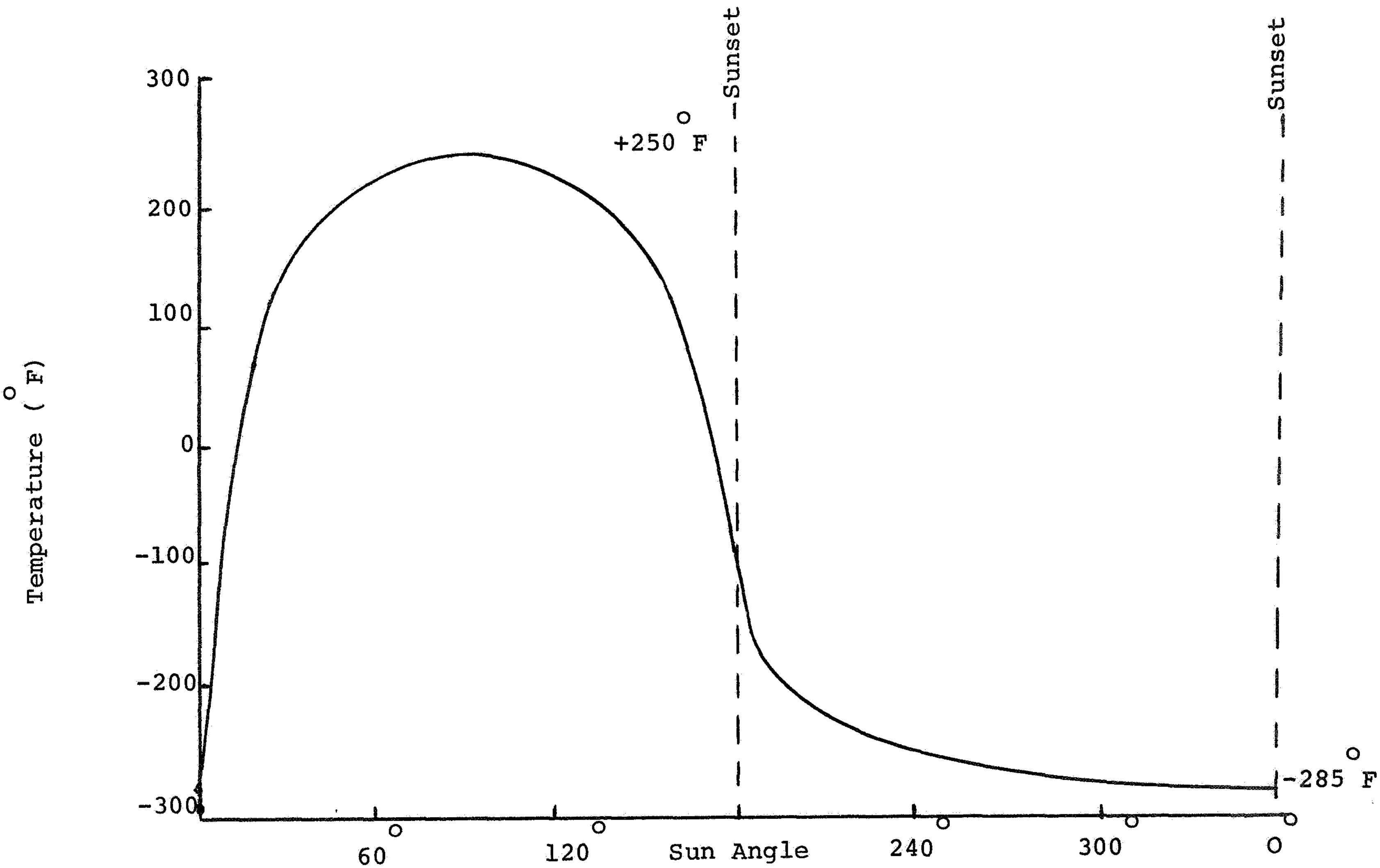


FIGURE 4.3-1 LUNAR SURFACE TEMPERATURE VS. SUN ANGLE

Another complication is caused by variable exposure to the sun due to shadowing. It can be 215 ^oF in full sunlight and -200 ^oF in adjacent areas of deep shade.[3] Lunar dust collecting on surfaces can cause additional problems since it causes an increase in the effective emissivity of surfaces if it is allowed to accumulate.

A plot of the lunar surface temperature versus sun angle from Malone, et al,[3] is shown in Figure 4.3-1. Design problems associated with the temperature extremes due to Lunar day and night and shadowing are:[1]

1. Distortion due to uneven temperature distribution
2. Expansion and contraction of parts
3. Thermal shock due to direct sunlight and shade cycling
4. Drastic changes in the mechanical properties of materials with temperature.

4.4 Lunar Terrain

For purposes of terrain classification, the lunar surface has been divided into two morphologic types, the mare and the upland. The mare regions are characterized by relatively gentle topography with low albedo. The upland regions have a higher albedo and are more rugged and complex. Adjectives are used in combinations with the morphologic names to further categorize the lunar surface. The U. S. Geological Survey have prepared models of smooth mare, rough mare, hummocky uplands, and rough uplands.[5] These are as follows:

Slope, Degree	Half Fraction of Total Slope Distance
Smooth Mare*	
0.20	0.05
0.40	0.05
0.60	0.05
0.80	0.05
1.10	0.05
1.30	0.05
1.60	0.05
2.10	0.05
3.00	0.05
3.80	0.025
4.80	0.010
Rough Mare*	
0.40	0.05
0.70	0.05
1.10	0.05
1.50	0.05
1.90	0.05
2.40	0.05
2.90	0.05
3.80	0.05
5.40	0.05
6.80	0.025
9.00	0.015
11.00	0.010
Hummocky Uplands*	
0.60	0.05
1.10	0.05
1.70	0.05
2.30	0.05
3.00	0.05
5.90	0.05
8.50	0.05
11.00	0.025
13.00	0.015
18.00	0.010

Rough Uplands*

0.80	0.05
1.50	0.05
2.30	0.05
3.00	0.05
4.00	0.05
5.00	0.05
6.00	0.05
8.00	0.05
11.00	0.05
14.00	0.025
18.00	0.015
23.00	0.01

*Base Line = 50 meters

Since most lunar regions are composite, the name given to any particular morphologic type merely reflects the predominant morphology of the region. A smooth mare region for instance may contain subregions which are rougher than some subregions in a rough mare.[4]

Terrain features are described by terms such as craters, domes, rilles, rays, ray systems, ridges, blocks, and faults. Such features are characterized by terms such as diameter, relief, rim height, block distribution, mean slope, roughness and many other terms.

Craters cover a significant portion of the lunar surface. Telescopic moon studies indicate that close to 10 percent of the lunar surface is covered by fairly large craters. In certain areas, over 20 percent of the local area is covered by craters. Graphs for numbers of craters (per unit area) larger than diameter D versus crater diameter are given by R.E. Hutton.[4]

Craters are caused by meteorite and comet impact. The impact results in certain block distributions around the rims of craters and between closely spaced craters. The reference by Hutton gives cummulative rock distributions in intercrater regions and the percent of area covered by blocks larger than a certain diameter D. Some topographical data for selected lunar regions is given in Table 4.4-1.

4.5 Lunar Soil

Chemical composition estimates of Lunar soil were made on Surveyors V, VI, and VII.[4] As on earth, the most common elements are oxygen, silicon, and aluminum with atomic percentages of about 60, 20, and 7 percent respectively. This composition corresponds to that of a basalt with a relatively high iron content (5%) for the two maria terrains and a relatively low iron content for the Highland terrain near Tycho (2%). Table 4.5-1 shows percentages of minerals and oxides found in samples taken from Apollo XI.

Data from both U.S. and U.S.S.R. spacecraft indicates the lunar surface is covered by a matrix of fine, somewhat cohesive particles less than 1 mm in diameter, with a few rocks scattered in and on the matrix.[4] There have been many estimates made regarding the thickness of the layer; Oberbeck and Quaide[6] stated that the layer varied from 1 to 20 meters. Scott[7] reported that the layer varied from 1 cm to 15 cm on an area checked by Surveyor VII. Jet Propulsion Laboratory Reports[4] indicate

TABLE 4.4-1 TOPOGRAPHICAL DATA FOR SELECTED LUNAR REGIONS*

Area or Feature	Slopes and Reliefs
Harbinger Mountains	Local Slopes of Rille Wall-39°
	Long Slopes of Rille Wall-22°
	Long Slope on Upland Ridge-13° to 19°
	Relief of Ridges - 200 to 400 meters
	Slopes on Small Rille Walls-15°
	Relief of Small Rilles-200 to 400 meters
Schroters Valley	Long Slope of Rille Wall-31° for 1 km
Near Aristarchus	Relief 663 meters Slope of Upper to Lower Plateau - 22° for 3 km
Small Crater in Schroters Valley (180 meter diameter)	Relief - 26 meters Slopes of Upper Walls to 21-31°

TABLE 4.5-1 MAJOR ELEMENTS AND ELEMENT OXIDES, APOLLO XI SAMPLES

Element	Range (%)	Oxide	Range (%)
K	0.053 - 0.18	SiO_2	36.0 - 45.0
Ca	6.4 - 8.6	Al_2O_3	7.7 - 13.0
Ti	4.2 - 7.5	TiO_2	7.0 - 12.5
Fe	12.1 -16.0	FeO	15.6 - 21.0
Mg	3.9 - 6.0	MgO	6.5 - 10.0
Al	4.0 - 6.9	CaO	9.0 - 12.0
Si	16.8 -21.0	Na_2O	0.20- 0.65
		K_2O	0.064- 0.22

*R.E. Hutton[4]

that the underlying material is a relatively strong rock material. Soil parameters for lunar soil are given in Table 4.5-2. These parameters show that the lunar soil has parameter values similar to terrestrial soils. Caution, however, must be exercised in using the values in equations which are valid on earth. It is impossible to assume that all terrestrial correlations will be valid on the moon.[8]

TABLE 4.5-2 LUNAR SOIL PARAMETERS*

Parameter	Value
Grain Size	2 to 60 Microns
Median	28 Microns
Cohesion	0.02 to 0.2 N/cm^2
Average	0.05 N/cm^2
Internal Friction Angle	31 - 39^{o}
Friction Co-efficient (Metal-Soil)	0.4 to 0.8
Adhesive Strength	0.0025 to 0.01 N/cm^2
Permeability	1×10^{-8} to $7 \times 10^{-8} cm^2$
Porosity	0.35 to 0.45
Bulk Density	0.7 to 2.5 gm/cm^3
Seismic Velocities	
Compression	30 to 90 m/sec
Shear	0.7 to 2.5 m/sec

*R.E. Hutton[4]

4.6 Lunar Lighting

There are three factors to consider in understanding the difference between lunar and earth lighting.[3] These are:

1. Illumination or incident light source difference.
2. The reflectance of the lunar surface compared to the earth surface.
3. Surface brightness as seen by observer.

Table 4.6-1 shows that the solar constant for the moon is 13.4 x 10^4 lumens/m^2 which is about 1.4 times the solar constant on the earth. This is mainly due to the lack of an atmosphere on the moon. The table also shows that full earthshine is greater than full moonshine by a factor of six. This is important since earthshine is the primary source of light during the new moon.

TABLE 4.6-1 ILLUMINATION CONSTANTS FOR EARTH AND MOON.

Units	Solar Constant	Full Earthshine	Full Moon on Earth
lumens/m^2	13.4 x 10^4	13.5	2.26
Foot Candles	1.2 x 10^4	1.25	0.021
cal/cm^2/min	2.0		

The brightness of a surface is a function of the illumination E, the albedo of the surface ρ and the photometric function ϕ.[4]

$$B = E \rho \phi$$

Albedo is generally defined as the percentage of the total illumination of a planet which is reflected from its surface. Local albedo is the ratio of a diffusing surface to the brightness of an absolutely white surface placed normal to the rays of the sun.

Values for albedo on various types of lunar surfaces are given in Table 4.6-2.

TABLE 4.6-2 LOCAL LUNAR ALBEDO*

Feature	Range of Values			Average
Mare	0.07	to	0.12	0.095
Upland	0.108	to	0.12	0.150
Entire Face	0.07	to	0.24	0.110
Back Side (Entire Face)				0.217

The photometric function ϕ depends on geometric relationships between the sun, the observor and one surface. Figure 4.6-1 defines the two angles which determine the function ϕ. The sun line and the viewing line define a plane called the phase plane which is perpendicular to the line of intersection of the phase plane and the lunar surface plane.

Figure 4.6-2 shows how the value of the photometric function ϕ can be obtained when α and g are know. The figure brings out some interesting points concerning vision on the moon. When the phase angle is small, the value of ϕ is high until the value of increases above 80° at which time it drops off rapidly. Such is the case when the sun is low and behind the observor. Another point is that as the phase angle increases, the photometric function decreases rather rapidly for a given angle of α.

*R.E. Hutton[4]

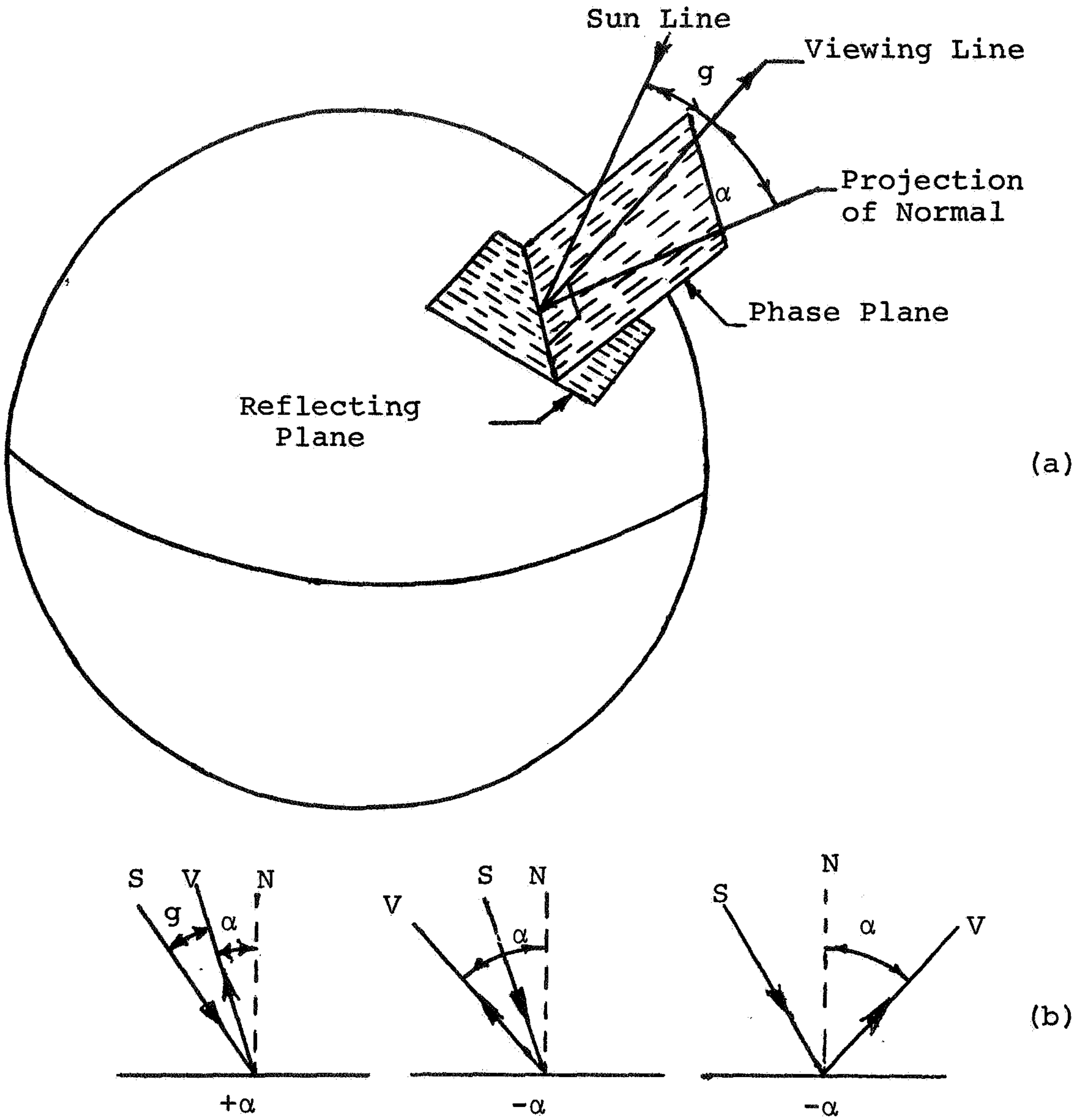


FIGURE 4.6-1 PHOTOMETRIC MODEL GEOMETRY

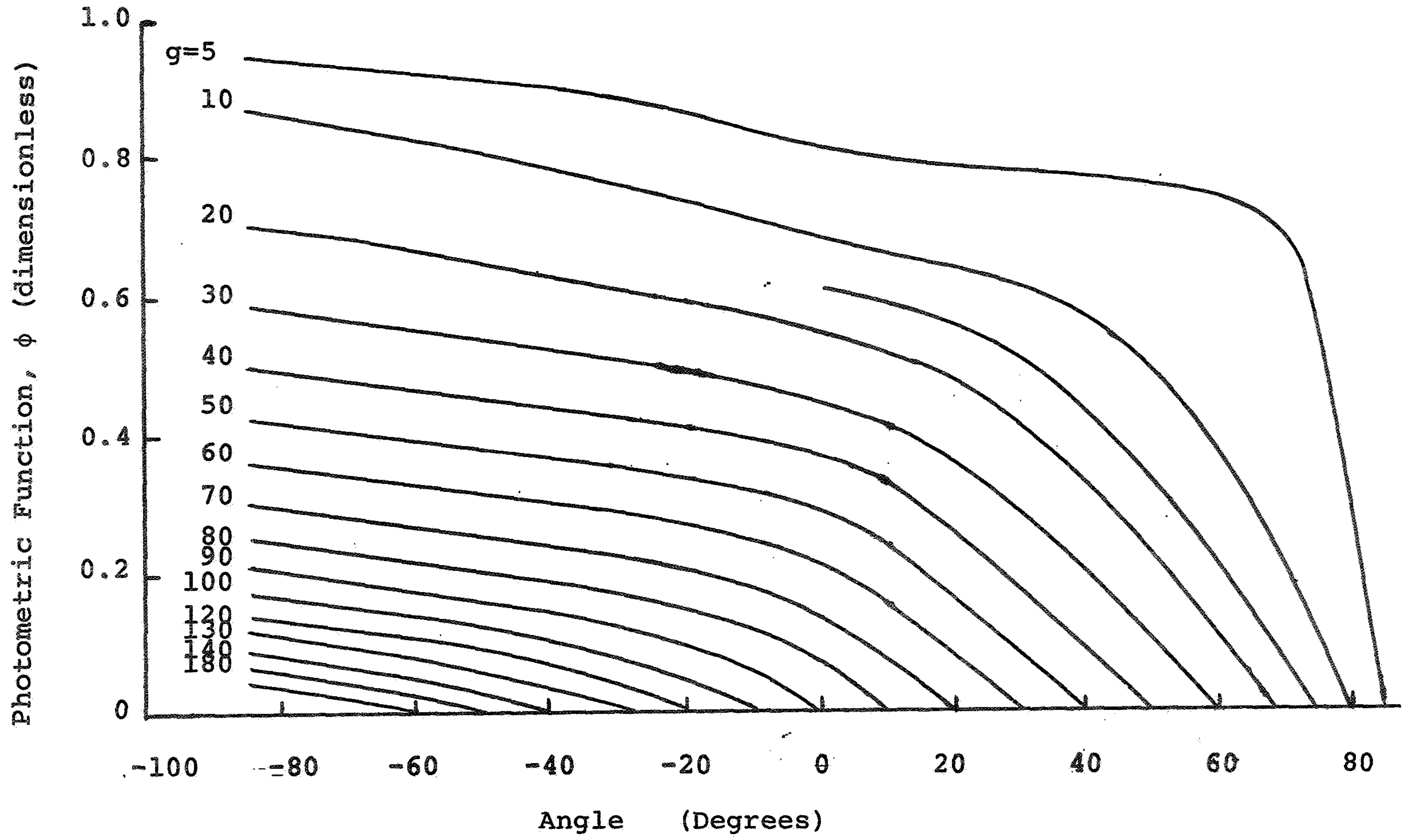


FIGURE 4.6-2 VARIATION OF PHOTOMETRIC FUNCTION WITH ANGLE α AND PHASE ANGLE (AFTER FEDORETZ)

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CHAPTER 5

GROSS SYSTEMS REQUIREMENTS

L. O. Degelman

5.1. Introduction

Basically, the establishment of the gross systems requirements for the lunar mobility aid was centered around one question, i.e., what will the mobility system be doing on the Moon in the 1980 period? To answer this question several sources of information were employed--namely, NASA's future plans for lunar exploration, project description documents by NASA, contractor documents describing several previously proposed lunar vehicles, and from lunar exploration concepts generated from within our design team. These concepts evolved into the establishment of several well defined mission tasks which would be required of a lunar mobility aid in the 1980's. Used as a basis for the gross systems requirements, these mission tasks provided the driving criteria which the mobility system would have to fulfill in order to be suitable for lunar use. This chapter summarizes those items of information which went into the establishment of the lunar missions and the resulting definition of overall mobility system requirements.

5.2. The Integrated Program Plan

Several of the key factors on which NASA has placed emphasis in recent planning are (1) commonality, (2) long lifetime, and (3) reusability. The importance of all these factors to the acceptance of any new designs was stressed to our design team by NASA in some early briefings at the Manned Spacecraft Center. In recognition of the first of these factors, i.e., commonality, NASA has set forth an integrated plan for future space exploration. This Integrated Program Plan (IPP) was described to our design team in several seminars and meetings during the first two weeks of the study. The IPP provided the basic guidelines and restrictions which were to be placed on the time and mode of delivery and the servicing of the proposed lunar mobility system.

According to the Integrated Program Plan, during and after the year 1973 an Earth Orbiting Space Station (EOSS) will be built in Earth Orbit and serviced by reusable shuttles which will travel between the Earth's surface and the space station. The Earth Orbit Shuttles (EOS) will be used to transport supplies and personnel back and forth between the Earth and the EOSS. A second type of vehicle, known as the Space Tug, will be employed to take supplies and personnel to lunar orbit to build another space station. This Lunar Orbiting Space Station (LOSS) is planned to be in a 60 mile polar orbit around the Moon.

Depending on future funding of the U. S. Space Program, large payloads for lunar exploration would be placed on the moon around 1982. This would be done by a lander version of the Space Tug. The payload on the Tug would consist of a mobility aid, crew and scientific experiment packages. Later, probably in the late 1980's or early 1990's, a Lunar Surface Base (LSB) would be constructed to carry out extensive lunar and Solar System exploration.

The space stations and the probable traffic pattern for the 1980's

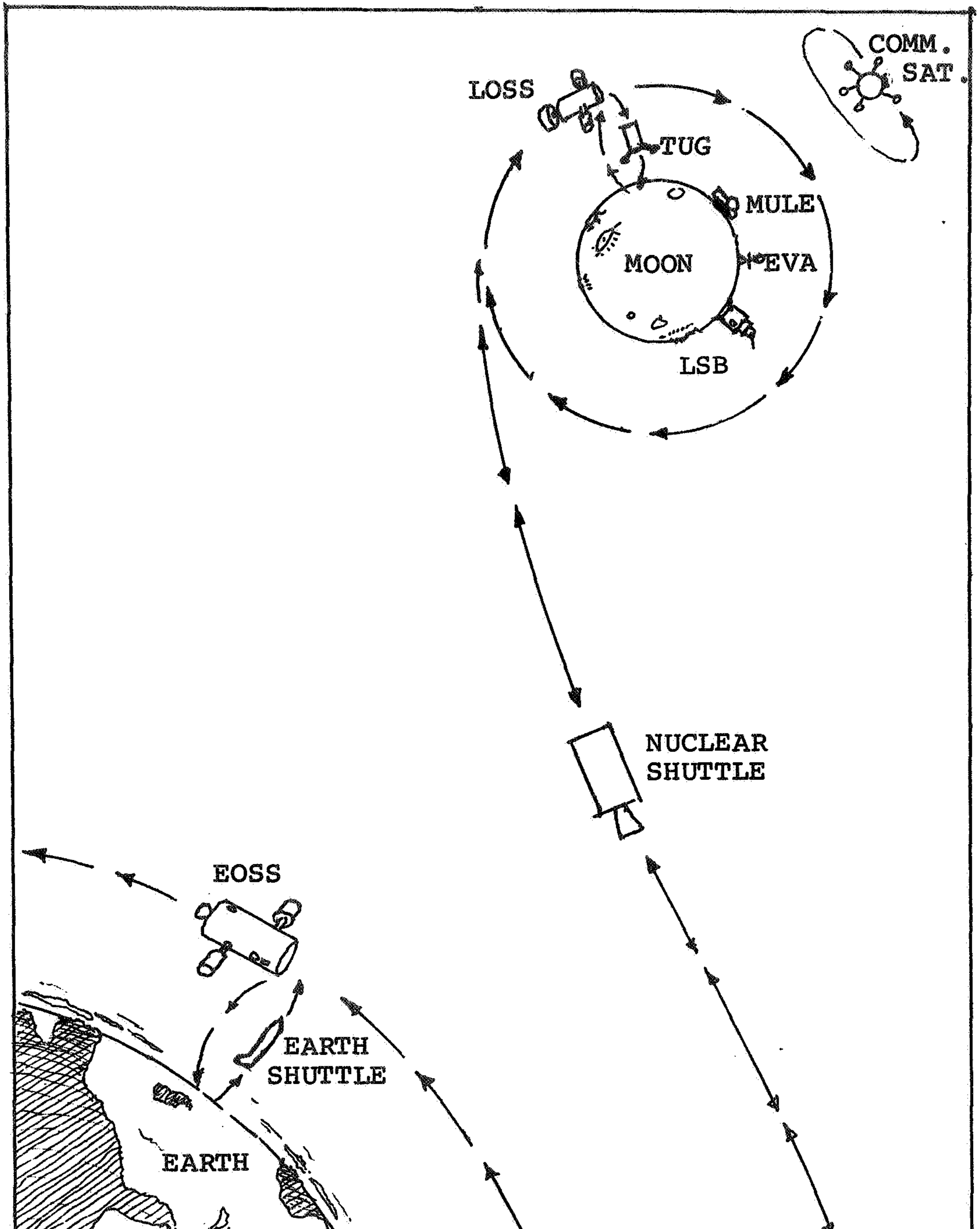


FIGURE 5.2-1. EARTH-MOON VEHICLE TRAFFIC PLAN FOR THE 1980'S

are depicted in Figure 5.2-1. Any large payload which reaches the lunar surface would first be put into Earth Orbit (possibly attached to the EOSS), transferred to lunar orbit to the LOSS, and finally landed via the Tug on the Moon's surface. The present IPP states that the lunar mobility system will be placed on the Moon by a lander version of the Tug.

What are the capabilities of the proposed Tug? Presently, it is planned that beginning around 1982 a Space Tug will land on the moon every three or four months. The Tug will contain a crew compartment capable of remaining on the moon for 28 days. The Tug will get its supplies for this 28 day stay period from the LOSS which will be in a lunar polar orbit.

The Tug will be able to soft land on the Moon in various configurations. Three of the typical configurations are shown below.

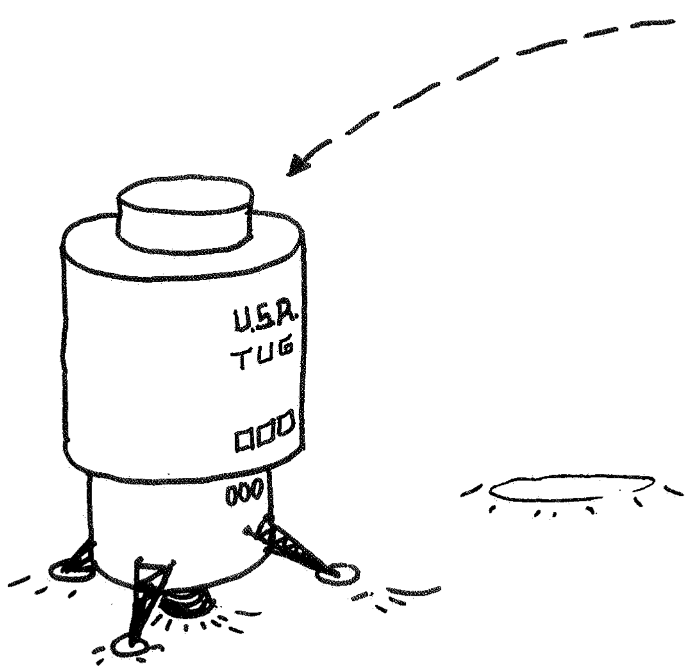


FIGURE 5.2-2. TUG WITH 70,000 LB. PAYLOAD

(1) One way unmanned--no return to orbit, the Tug can deliver 70,000 pounds to the lunar surface. This typifies the method of establishing a fairly substantial Lunar Base fully erected prior to landing.

(2) Unmanned, the Tug can deliver a 50,000 pound payload to the surface and return the propulsion unit (weighing around 20,000 pounds) to orbit.

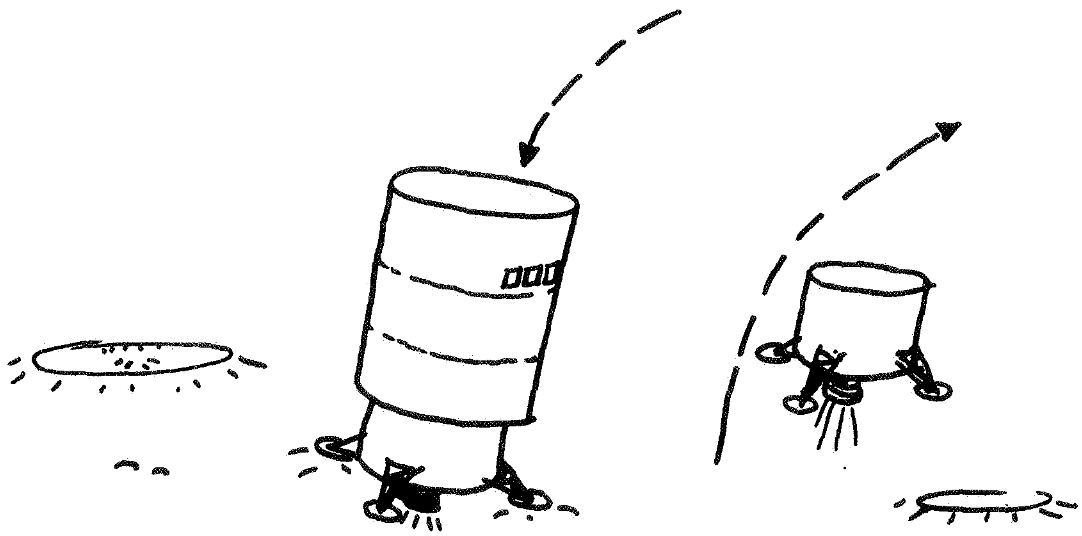


FIGURE 5.2-3 TUG WITH 50,000 LB. PAYLOAD

(3) Manned, the Tug can land 20,000 pounds of payload and return 20,000 pounds to orbit.

In the third mode of landing, i.e., the manned mode, the payload can be separated into 10,000 pounds for a 4-man crew compartment and 10,000 pounds of scientific payload. On the return trip to the LOSS it is reasonable to assume that a substantial amount of lunar samples could be returned, their weight being substituted for certain scientific experiment packages which would be left permanently on the lunar surface.

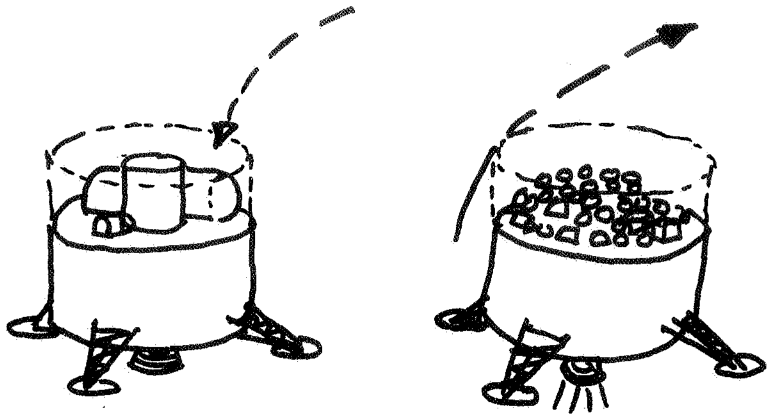


FIGURE 5.2-4 TUG WITH 20,000 LB. PAYLOAD

Generally, we can conclude from the 10,000 pound payload limitation of the Tug in the manned mode, that our mobility aid (including its fuel and scientific instruments) cannot exceed 10,000 pounds. This fact imposes an immediate restriction on the general class of mobility vehicle and the amount of time that the vehicle can spend traversing the surface of the Moon. We considered several classes of mobility systems in this study, but some had to be eliminated because of this weight limitation. These candidate systems and their elimination are discussed in Chapters 7 and 8.

A second restriction on the design of a lunar mobility aid pertains to size. The payload compartment of the Earth Orbit Shuttle is 14.5 feet in diameter by about 30 feet long. The diameter of the Tug lander is around 22.5 feet. Since the mobility aid must be transported from Earth to the Moon via the Shuttle and the Tug it must be configured to fit into a 14.5-foot diameter tube and must be able to be placed on the Tug's 22.5-foot diameter surface while landing.

Thus, we see that the mobility system is restrained in weight and size by virtue of the Integrated Program Plan. Presuming that time will also be a factor in the design we can also observe the time restraints involved with the IPP. The Plan calls for Tug landings at 3 or 4 month intervals, the first 7 of which will be used for lunar exploration and selection of a site for the first Lunar Surface Base. The LSB will be established on or about the eighth Tug landing. Because of the necessity to rotate crew at the Base, supply fuel and food, and to return specimens of scientific interest to Earth, a Tug will land every two months to service the LSB. We have assumed that certain Tug landers will still be required to place mobility aids on the lunar surface, even after a LSB is established. These landers would be in addition to those required to support the LSB. A probable schedule of this Tug landing activity is shown at the end of this chapter. (See Section 5.9)

The IPP, though tentative in nature, had a bearing on all the areas of considerations in defining the gross systems requirements. Reference will be made from time to time throughout the remaining portions of this chapter to the IPP as it relates to the topic being discussed.

5.3. Mobility System Activities

This section discusses generally what type of activity the mobility system will be required to do on the lunar surface, but does not go into detail in any one area. The details of the lunar missions are described later in this chapter.

The first seven Tug landings are exploratory in nature, but the stay times will be much longer than the previous Apollo missions (28 days vs. 3 days). Therefore, there is at least an order of magnitude increase in staytime that will be provided. Besides extending the staytime, a much larger area of the Moon will be able to be explored when the Tug lander is operational. Even though the range of the proposed Apollo Lunar Rover Vehicle (LRV) is about 80 km., the exploration radius is limited to 5 km. from the LM imposed by the walkback capability of the astronauts in case of a failure in the LRV.

Assuming that a mobility system can be designed to extend the range of the astronaut from his landing point, then a much more extensive lunar exploration program can be conducted. This should provide an increased overall value of lunar exploration. Lunar exploration, far in excess of that of the Apollo program, thus becomes one of the most required activities of the mobility system under consideration.

A second requirement of the mobility system would be that it support the LSB, once established, and possibly even in establishing the base. One activity would be simply transporting personnel and supplies from a landed Tug to the LSB and back again.

A third role that a mobility system on the Moon can play is one of research. The mobility device can be used to research and develop new mobility concepts for future planetary missions.

Summarized, these activities are:

1. Conduct Long Range Lunar Exploration.
2. Support Science Experiments.
3. Aid in Establishing a Lunar Base.
4. Support and Supply a Lunar Base.
5. Develop Mobility Capability for Planetary Missions.

Since the lunar exploration is earliest in the plan for the mobility aid and has some fairly definable objectives, items 1 and 2 in the above list were assigned prime importance. Lunar exploration is therefore discussed in further detail in the following section.

Establishing and supplying a lunar base is a little further in the future than the lunar exploration and has some uncertainties connected with it; therefore, it received secondary consideration. The last item on the list seems to come about naturally, since any use of the mobility system will tend to reveal certain good and bad features in its operation. The findings will provide valuable inputs to the design and use of vehicles on other planets.

5.4. Lunar Exploration

The lunar exploration program which began with the isolated surface observations of Surveyor and Apollo missions will evolve into an attempt to tie these isolated sites together into a coherent network. This will be accomplished through overhead orbital and Earth-based surveys to provide the essential mapping and remote-sensing data. The exploration program is expected to expand in the 1980's to rather long term surface explorations. These missions (according to References 1, 2 and 3) will consist of both on-site experiments and long traverse experiments and observations.

What are some of the experiments to be conducted on the Moon -- or why are we exploring the Moon at all? Without dreaming about health resorts on the Moon or exotic sports and games, the National Academy of Sciences has recorded 15 basic questions about the Moon for scientific purposes only. These questions become the basis on which the lunar experiments and equipment will be planned. These 15 basic questions are [4]:

1. What is the internal structure of the moon?
2. What is the moon's geometric shape?
3. Internal Energy? Seismically active? Active volcanism? Internally produced magnetic field?
4. Composition of surface rocks?
5. What is responsible for relief of lunar surface?
6. What is the moon's tectonic pattern?
7. What is the process of erosion of the Lunar surface?
8. Volatile substances on the surface?
9. Organic materials?

10. Age of moon?

11. History of the moon-earth interaction?

12. Thermal history of the moon?

13. What is the flux of solid objects striking the surface?

14. What is the flux of cosmic radiation and solar radiation at the lunar surface?

15. Are there magnetic fields in the lunar rocks?

Based on these questions a lunar exploration program has been formulated which will enable scientists to answer these questions. Basically this lunar exploration program will consist of:

1. Direct observations and measurements by scientist-astronauts carrying instruments and tools.
2. Sample collection for return to Earth.
3. Emplacement of long-lived instrumentation.
4. Subsurface drilling and sampling.

Some of the scientific exploration experiments which have been proposed in order to accomplish the lunar exploration program consist of the following:

A. Lunar Atmosphere
   1. Total Atmospheric Pressure
   2. Atmospheric Composition
   3. Neutral Gas Composition
B. Geodesy
   1. Selenodetic Surface Surveys for Ground Control for Orbital Mapping
   2. Surface Gravity

3. Seismic Measurement -- involves placement of Seismometers and Discharging Explosives

4. Earth-Moon distance

C. Geology

1. Lunar Sample Collection

2. Subsurface Structure -- involves drilling of deep holes for core collection and heat flow probes.

D. Geophysics

1. Seismic Study of Lunar Materials

2. Lunar Seismic Events and Tectonic Measurements

3. Study of Meteoroids in Lunar Environment

4. Lunar Surface and Material Hardness

5. Electromagnetic Properties

6. Electric and Magnetic Field Surveys

7. Study of Lunar Materials with Nuclear Physics Techniques

8. Lunar Surface and Sub-Surface Temperatures -- involves deep hole drilling for temperature probes.

E. Geochemistry

1. Analytical Chemical Analysis

2. Radionuclides Present on Lunar Surface?

3. Thermal Analysis Investigation

4. Chemical Analysis for Lunar Gases

F. Particles and Fields

1. Solar Charged Particles at Lunar Surface?

2. Solar Wind Particles at Lunar Surface

3. Magnetic Field Strength and Time Variations

4. Electric Field Strength and Time Variations

G. Biology

1. Prebiotic Chemistry and Evidence of Existing Life in Lunar Materials.

What will be the function of a mobility aid in the conduct of these lunar surface experiments? The basic functions of the mobility aid will be:

1. To transport instruments, collect and identify samples, telemeter observations to Earth, LOSS, or the LSB.

2. Emplace instrumentation along the route of a traverse, or alternatively to pause at selected points long enough to permit stationary experiments.

3. To explore unknown sites and targets of opportunity, and to evaluate and compare sites for future manned investigations in greater depth, or other special purposes such as astronomy or recovery of Lunar water.

The lunar mobility system should have the capability for performing long-range geological and geophysical traverses by remote control, making scientific measurements with on-board instrumentation, deploying small self-contained packages of geophysical instruments, and collecting lunar samples from widely separated areas of the moon for delivery to a rendezvous point with a later manned mission.

The priority goal of the long traverse is the collection of lunar material samples for eventual return to earth. These samples would be identified at the sampling site and returned selectively. Therefore, there is a need to have instruments aboard the mobility aid which will permit selection or rejection of samples, to record all significant aspects of the surroundings, and to detect anomalous conditions that might require a change in mission strategy.

For geology, geophysics and geochemistry, travel across large regional units of the Moon and visits to distant features are desired, meaning that traverse dimensions should be comparable to those of maria and highlands, namely several hundred kilometers. The geophysical experiments involving seismic and gravimetric measurements will require baselines of 1000 km. or more.

Since these traverse distances are relatively large, the mobility system should have a range-independent prime power supply, and it must routinely survive the lunar night. Lifetime to wearout then becomes a dominant design consideration rather than range.

A couple of ways to operate a mobility aid under the dual-operating mode have been suggested[3]:

1. The vehicle(s) and men are delivered to the lunar surface aboard a Lunar Tug; the vehicle is deployed, put into the unmanned automatic mode, and sent off on a long traverse. The crew is free at the landing site to do local experiments.

2. Vehicle is delivered aboard on unmanned Tug some time ahead of men, and the traverse is completed under remote control. A rendezvous is made with the vehicle when the traverse is complete, the vehicle is put into manual mode and used by men on shorter

traverses around the landing site.

The Mimosa reports[2,3] go into some specific detail on the conduct of the lunar exploration patterns, namely:

1. Tug lands at some predetermined point on the lunar surface for a 28-day stay.

2. Local investigations can be conducted by spacesuited astronauts.

3. With some mobility aids a 14-day surface investigation would be conducted over an area which far surpasses that of the Apollo program.

4. Tug leaves the surface and returns three months later to a second site. Here, the Tug might rendezvous with the mobility aids which were deployed at the first Tug site. The program then repeats steps 2, 3, and 4.

This exploration sequence is depicted in Figure 5.4-1. The figure shows several Tug landing sites representing the approximate scale of the exploration program to be conducted in a one year period. The approximate ranges of expected exploration are shown in Figure 5.4-1 as the circles around each Tug site. We could note, however, that a Tug may be required to land at a given site more than once because of the exploration requirements.

The types of manned traverses which would be expected in the areas of the Tug landers are indicated in Figure 5.4-2. In these traverses scientific packages would be deployed along the way, lunar samples would be collected, and continuous data would be recorded by onboard sensors.

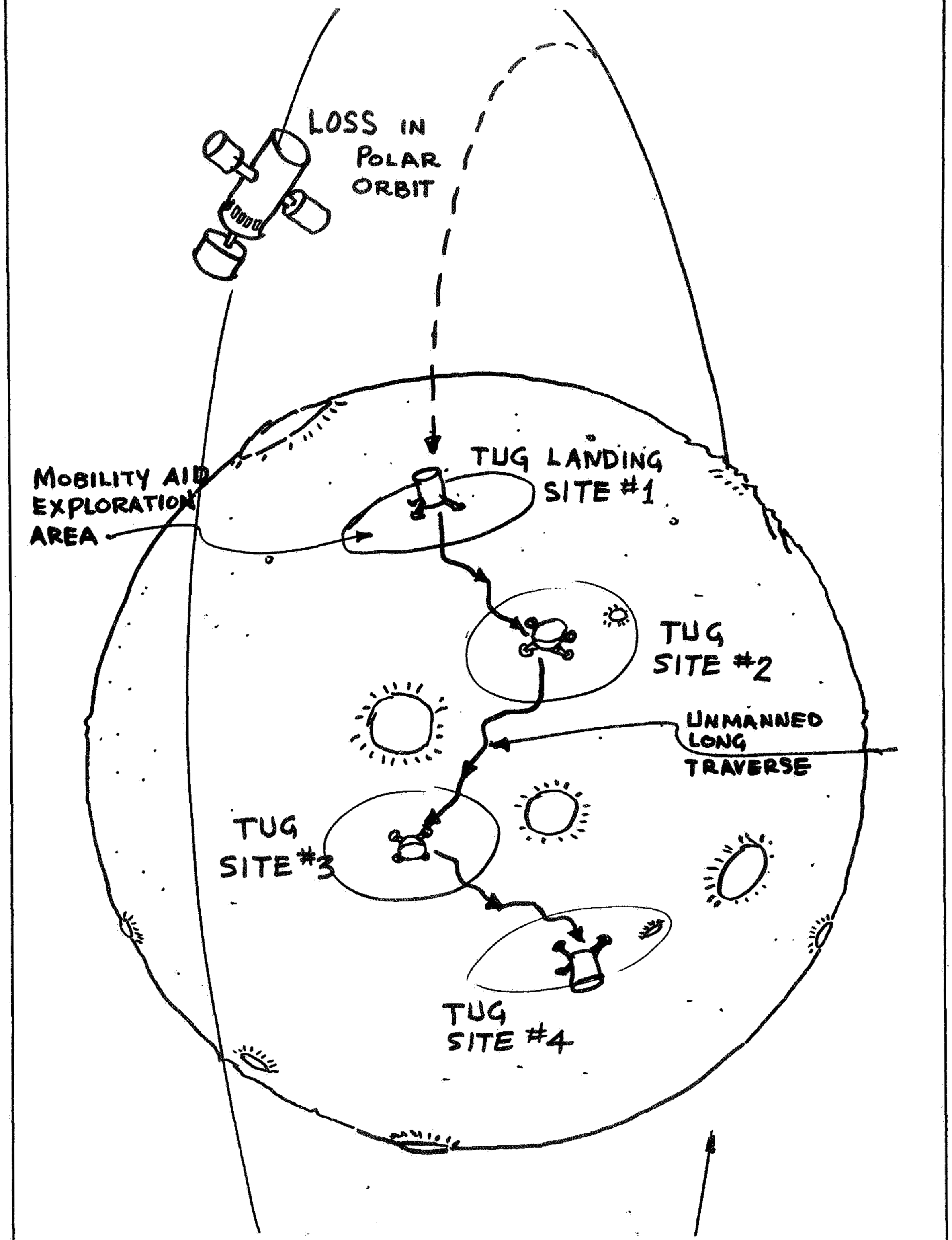


FIGURE 5.4-1 LUNAR EXPLORATION PATTERN

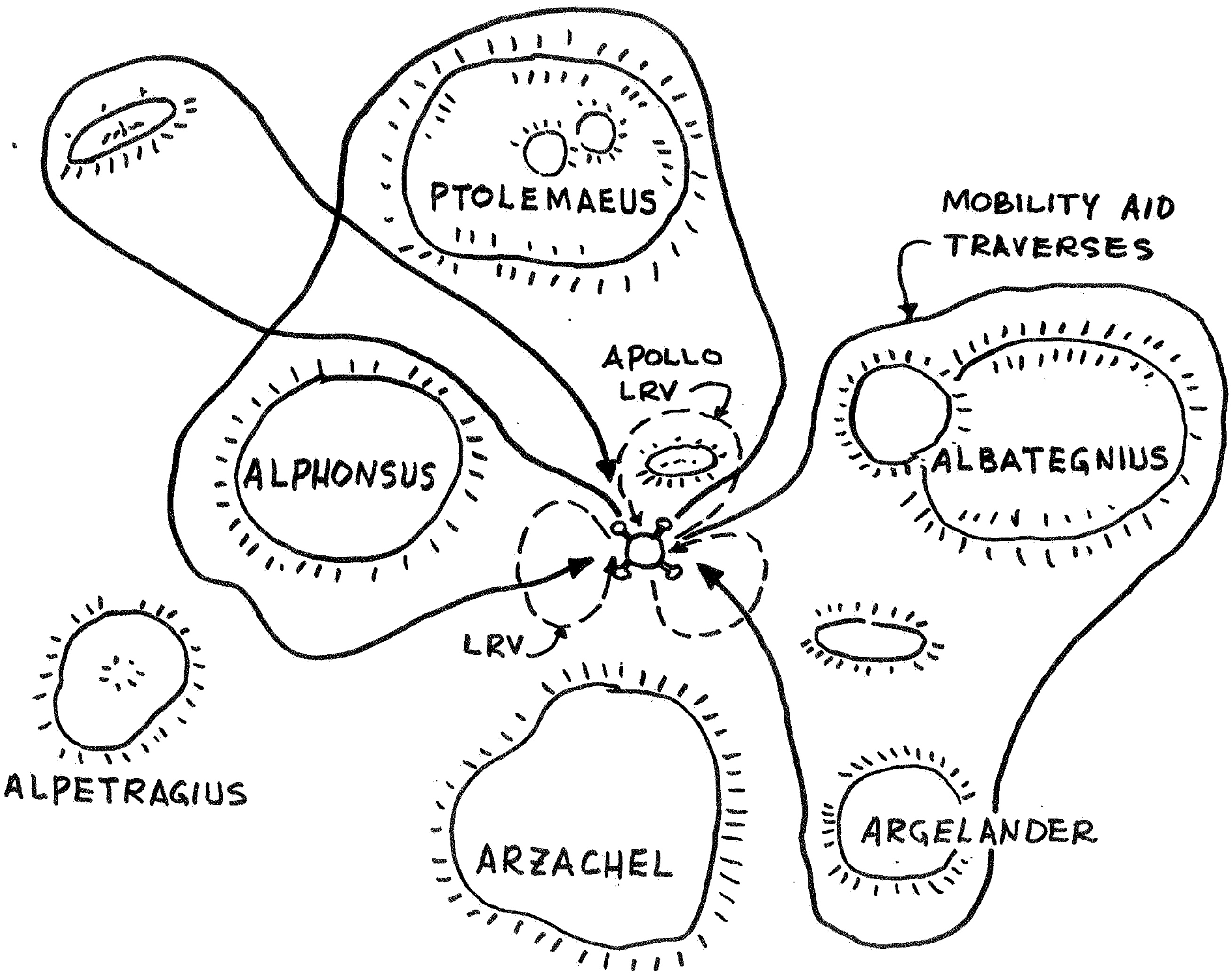


FIGURE 5.4-2 <u>TYPICAL MOBILITY AID TRAVERSES AT LANDING SITE 14° SOUTH OF MOON'S EQUATOR</u>

It has been determined[5,6] that the mobility aids which are employed for surface explorations -- whether manned or automated -- will be monitored and/or controlled from the Lunar Tug, and that the first seven Tug landings will be primarily for support of lunar explorations.

The early surface missions of 14-day durations will be conducted by four crewmen who will employ the "buddy system" of exploration, with a two-astronaut team performing the surface exploration.

Portions of the surface EVA's will be devoted to emplacing "Science Stations" which will continue to transmit scientific data after the astronauts leave the site. In addition to the emplacement of science stations the astronauts will also conduct "locale" experiments in the vicinity of the Tug, including the collection of surface and subsurface samples. These samples will be analyzed to the degree possible within the Tug, with complete analysis being reserved for the LOSS and Earth Laboratories.

According to the first five references the planned experiments which are to be conducted on the various missions are fairly well established. We used these experiment requirements, therefore, to set forth the traverse requirements for the initial exploratory use of the lunar mobility aid. One of the typical traverse patterns is shown in Figure 5.4-3. The sketch indicates two medium range manned traverses and one long unmanned traverse. On the first traverse a science station or two would be deployed as would the deep drill. This deep drill is to drill a 100-foot deep hole

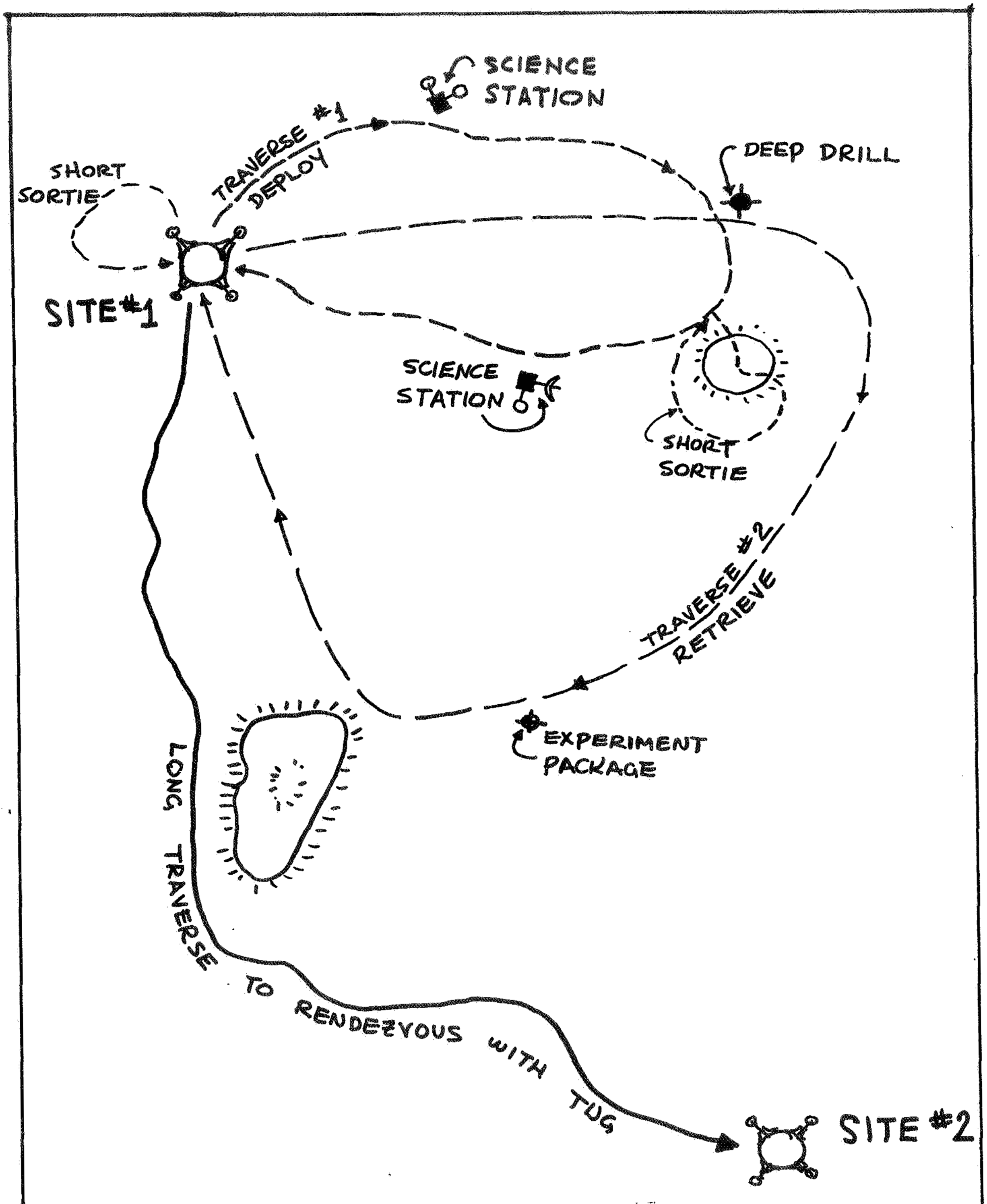


FIGURE 5.4-3

MOBILITY AID TRAVERSES BETWEEN TUG SITES and collect twenty 5-foot core samples. Since the drilling operation itself would take about 60 hours[4], the drill would be left in place and retrieved with the core samples on the second manned traverse. After a staytime of 28 days the Tug would depart and the mobility aid would be sent off on the long unmanned traverse. The mobility aid would then rendezvous with the Tug on a subsequent landing. The experiments which would be conducted on the unmanned traverse are described in Reference 4. These experiments defined the basic requirements for the unmanned missions; these are summarized in part 5.8, the last section of this chapter.

5.5. Support of Lunar Base

Late in 1983 lunar exploration utilizing the manned Space Tug and surface mobility aids should be into its seventh mission. This represents a period of about 20 months from the first Tug landing. The eighth Tug landing is scheduled to deliver the first Lunar Surface base. The LSB will be a 70,000 pound derivative of the basic EOSS and LOSS modules according to the Integrated Program Plan. The LSB will be unmanned upon delivery and will utilize a Tug which is scheduled for disposal. Probably early in 1984 the ninth Tug landing will deliver to the Lunar base a crew of four astronauts who will man the LSB. Using this basic plan, a mobility aid mission requirement denoted as "Lunar Surface Base Support" should be defined based on the IPP.

It is conceivable that the LSB will require deployment in an area that is not as flat as desirable. Therefore, the mobility aid which lands with the seventh Tug lander could act as a bulldozer to smooth out the desired locale of the LSB.

Since the crew of the LSB will be delivered by Tug landing #9 and the Tug cannot land too close to the LSB, the mobility aid could act as a personnel vehicle to transport the crew between Tug Lander #9 and the LSB. A crew rotation is envisioned to occur every four months. From the time the LSB is manned and activated until its lifetime is complete it will require periodic refurbishing of personnel and supplies. These supplies will be delivered to the Lunar surface by the Tug every two months. Assuming that the Tug cannot land too close to the LSB, a means of transport is required between the Tug and LSB. This can be accomplished by the mobility aid.

A LSB will require many Tug landings in close proximity during its lifetime. It will be necessary, therefore, to construct a convenient permanent landing site and various landing aids for these many Tug landings.

Finally, the mobility aids could also be used to assist in providing a protective covering for the LSB. This would involve coating the exterior of the LSB with moon dust or some other substance which would reduce the degrading effects of "Lunar day-Lunar night" temperature extremes, micrometeorites and high energy particle collisions. Assuming that the presence of man on the lunar surface is going to be a long term affair, a permanent base will have to be constructed. The LSB which is delivered by a spent Tug can be considered as temporary in nature. Being delivered fully constructed, having limitation on crew number, and requiring 100% outside supplying cannot be considered as being the ideal lunar base configuration.

In order to make the lunar staytime reasonable and economical a lunar base should be constructed with emphasis placed on utilizing all that the moon can provide in terms of supplies, permanent transportation techniques between landing vehicles and lunar base should be developed, such considerations as plant farms for food and oxygen supply would be reasonable, large banks of solar panels could be deployed for acquiring electrical energy. Remote power generation utilizing nuclear energy with transmission lines to the lunar base is a possibility. According to NASA, these activities are more than likely delegated to the 1990 period or later. At the same time, with this permanent lunar base in operation, lunar and solar exploration would be continuing.

The requirements of mobility aids in this context would be;

1. Hauling supplies and personnel between the tug and lunar base.
2. Lifting or erecting.
3. Leveling and digging.
4. Drilling
5. Cutting and welding
6. Cable laying.

Since we have to speculate on the lunar missions in the late 1980's these requirements (activities) will have to be regarded as secondary to the main function of the mobility aid, there will, most likely, be a second generation mobility aid required for the 1990's based on the current requirements.

5.6. Mobility System Constraints and Guidelines by NASA

At the beginning of the project NASA issued a "Statement of Work" which contained the guidelines and assumptions which were to be used in the project. Several slight modifications were made by NASA in the first couple weeks of the program resulting in the following requirements:

1. The mobility system concept developed shall satisfy gross mobility system requirements consistent with currently envisioned Integrated Program Plan goals and objectives of lunar operations for the post 1980 period.

2. The range for total distance traveled during an extended operation shall be up to 250 km. for manned activities and shall be up to 1500 km for unmanned activities.

3. The life support capability shall be such that the mobility system will continuously support two men for at least 36 hours with 12 hours of contingencies when man is involved as an integral part of the activity being performed by the mobility system.

4. The payload of the mobility system may include at least one or both of the following:

   (a) Men, 200 lb/man for shirtsleeve activities
   400 lb/man for EVA activities

(b) Science packages, 1000 to 2000 pounds per sortie.

5. The operational capability of the mobility system shall extend to lunar night activities with minimum reduction in performance.

6. The lifetime of the mobility system shall be at least one year.

7. The crew size of the mobility system will be two.

8. The dry weight of the mobility system shall not exceed 5000 lb.

9. The delivery of the mobility system to the lunar surface shall be by a lander version of the Space Tug.

10. The activities of the mobility system shall include extended lunar exploration and support of lunar base construction.

11. The capabilities of the mobility system shall include the following:

(a) Climing and descending slopes of 30 degrees

(b) Ground clearance of 50 cm.

(c) Obstacle negotiation of 50 cm for a step and 90 cm for a crevasse.

(d) Stability at pitch and roll angles of 45 degrees.

12. The completion of mobility system activities shall be provided when a failure occurs in a subsystem.

13. A lunar orbiting space station shall be in existence in lunar orbit at the time of manned surface operations.

14. Surface logistics shall be delivered by a lander version of the Space Tug.

15. The lunar soil is specified per "MSC Specification Lunar Soil Model" (preliminary), March 10, 1970.

16. The lunar surface roughness is defined in "Lunar Surface Models" by R. E. Hutton, March 1969, TRW Document No. 11468-6001-R_O-00.

5.7. Mobility System Guidelines by Design Disciplines

Each group from our design team also issued requirements which were to have a bearing on the outcome of the final mobility concept. These requirements and guidelines are included in this section under each group heading.

5.7.1. Power and Propulsion

1. Gross weight with fuel must be less than 10,000 pounds for a normal Tug delivery.
2. The normal fuel delivery would be 5,000 pounds or less
3. NASA specified that if chemical propulsion were used, Oxygen must be used as the oxidizer. This maintains commonality with the Tug propulsion system which uses liquid oxygen and liquid hydrogen, and excess Tug oxidizer could be used on the mobility aid if necessary. It was also suggested that hydrogen be considered as the prime fuel.
4. It was found through investigation of literature on lunar surface characteristics that surface objects around the size of 1.2 meters would cause most of the vehicle immobilization. Obstacles larger than that could be predicted earlier and thus avoided, and obstacles much smaller than that size could conceivably be crossed over by the vehicle.
5. Power subsystems types to consider include (a) reaction jets and batteries for flyers, (b) reaction jets, gas generators,

and batteries for ground effects machines (GEM's) and (c) fuel cells, radioisotope thermoelectric generators (RTG's) for roving vehicle.

6. The normal to maximum speeds of the ground restrained vehicles on the moon are as follows:

(a) 5 to 16 km/hr for rovers.

(b) 10 to 32 km/hr for GEM's.

7. The deep core drill will require the most power of all the scientific instruments, i.e., about 3 kilowatts.

5.7.2. Astrionics

1. Because the mobility system is to operate by remote control on unmanned traverses it must be monitored constantly. It must also be capable of operating on the far side as well as the near side of the moon. Because of this a satellite system must be established which will provide communication links between all points in the network. This involves communication among the mobility system, Tug, LSB, LOSS, and Earth. Several possible satellite configurations were investigated, including the Equatorial, the Polar, and the Libration Point Satellites. The conclusion reached was that the Libration Point types were the most feasible ones to consider for this use and for other lunar exploration which might be conducted. Details of these satellite systems are discussed in Chapter 19.

2. The mobility system could be controlled from the LSB, LOSS, or the Earth-based manned spaceflight network. Primary

control is to come from the LOSS until a base is established at which time the LSB should be considered for the control point.

3. In regard to transmission of scientific data the mobility vehicle will act as a data transmission center for on-site experiments while the experimental packages and emplaced science stations would be centers of transmission to the LOSS or Earth.

4. The communication links must be designed to withstand a single point failure between any two stations of the network.

5. In the unmanned mode there must be an automatic obstacle avoidance system and two TV cameras on board the mobility vehicle. These are required since transmission delays between the vehicle and the control base could make the remote control aspect ineffective in avoiding very close obstacles and crevasses. Even in the manned mode there must be an obstacle warning system since the solar illumination on the lunar surface can cause loss of depth perception for the astronauts at certain viewing angles.

6. The tracking system must be designed to locate a fixed point on the Moon within 10 meters. This is essential because it is necessary to know where certain scientific instruments are placed for transmission of data back to Earth. For the moving vehicle it is necessary to locate it to an accuracy of about 500 meters. It was reasoned that if the vehicle could be positioned (using tracking data) to within 500 meters of a certain destination point it could be steered visually to its target by use of the TV cameras or by the onboard crew if manned.

7. The mobility vehicle must have an onboard general purpose digital computer. It's uses would be for data processing, naviga-

tion, control, and certain experiment calculations and data reduction.

5.7.3. Human Factors

1. There must be a pressurized suit on board the vehicle for each crew member for emergency reasons.

2. Closed cabin/shelter weights are in the vicinity of 2000 pounds for a vehicle cabin and 400 pounds for an inflatable space shelter.

3. There must be thermal protection; thermal control system; and life support capabilities; such as food, oxygen and sanitary facilities on board the vehicle.

4. A work day cycle using the mobility system is to be considered as 18 hours long - 8 hours of work, 3½ hours for communications, eating and housekeeping, and 6½ hours for sleeping.

5. There must be teleoperators attached to the exterior of the mobility vehicle which are operated by the astronaut inside the crew cabin. This provides safer working conditions for the crew, and since the crew can work in a shirtsleeve environment it also provides for more comfort. In addition, the astronauts heat output is less and thus there is less load on the environmental control system.

6. The automatic guidance system is necessary for the reasons stated in the previous section, i.e., that the astronaut's depth perception of hazardous objects is sometimes lost when the sun angle and viewing angle are close to each other.

5.7.4 Configurations

1. The mobility system will be transported to lunar orbit in a compartment 14.5 feet in diameter by 30 feet long. It will then be landed on the lunar surface atop a Space Tug with an approximate diameter of 22.5 feet. Therefore, the vehicle must be designed to fit into a 14.5-foot diameter tube and be on the order of 22 feet long or less.

2. Inactive storage on Earth prior to launch should be considered.

3. In some cases the mobility system will be required to return sealed and/or refrigerated lunar samples back to Earth.

4. The mobility system must be configured in a modular fashion to permit: science station emplacement, deep drilling, sample collection, crew transport and life support, teleoperator operation, control and guidance, and transport of fuel.

5.8. Definition of Candidate Missions

After the NASA restraints, requirements, group guidelines, and reference materials were reviewed it became clear that the mobility system would be involved with many varied tasks. These tasks were outlined and organized into groups which would help us define exactly what the vehicle would be doing on the Moon and how often it would have to perform each task. The grouping of these various tasks formulated the ten candidate missions which are outlined in this section.

After the ten missions were outlined, different weights were assigned to the missions based on their judged relative importance to the objectives of being on the Moon. These weights were used in a later study which showed the adequacy of various candidate mobility systems in performing all ten missions. In rating the adequacy of candidate mobility systems by this method it was possible to eliminate many from further consideration based on a low overall effectiveness. This technique is discussed further in the "Cost-Effectiveness Study" in Chapter 9.

The document entitled "Traverse Science Data Package"[4] seems to give the clearest definition of the tasks and devices which are to be involved with the lunar exploration missions. The experiments defined in the document are all directed toward answering the 15 fundamental questions that were posed by the National Academy of Science. Of special interest is the apparent emphasis on the requirement to employ a deep drill on the lunar surface. The following quote, taken directly from the referenced paper, is in reference to a deep drill (30 meters) and core samples:

"Core drilling is regarded as one of the most important means of acquiring information on the structure, composition, history, and internal processes of the Moon. Several scientific experiments important to the pursuit of basic knowledge about the Moon are associated with depth drilling. Major groupings are as follows:

1. Acquistion of undistrubed subsurface samples
2. Measurement of subsurface characteristics

The deep drill is an adaptation of the wireline drill which has been in widespread use on earth for the past 18 years. The wireline drill system consists of a drillrig, or prime mover, drill rods, a core barrel and a bit, and employs conventional rotary drilling. As the core is cut, it is stored in a retractable core barrel which can be lifted to the surface without withdrawing and dismantling the drill rods and leaves the bit in place. After a length of core is withdrawn, an alternate retractable core barrel can be lowered into place inside the drill rod and drilling continued. Equipment is automated to enable most advantageous use of astronaut time.

In 1968 a model of the lunar drill had been developed by Westinghouse Defense and Space Center. Two drill systems were fabricated and delivered to Marshall Space Flight Center. At that time modifications were being considered to make the system automated.

It is planned that the deep drill be used on both the manned and unmanned traverses. The weight of the drill with automation unit, batteries, and power conditioning is around 1030 pounds and occupies a volume of 6 cubic feet. The 20 sample containers (each 2.5" dia. by 5 ft. long) weigh 40 pounds total and will occupy 10 cubic feet of space. The core samples will weigh an additional 200 pounds; therefore, the use of the deep drill means transport of 1270 pounds and a volume requirement of 16 cu. ft. for storage.

Other experiments will weigh a total of 500 pounds and occupy a volume of about 20 cubic feet. When "science stations" are transported, their weight should be between 500 and 1,000 pounds. Several combinations of drill, experiments, and stations are put together in the missions defined in this section as can be seen from the weights indicated.

5.8.1. Unmanned Long Traverse

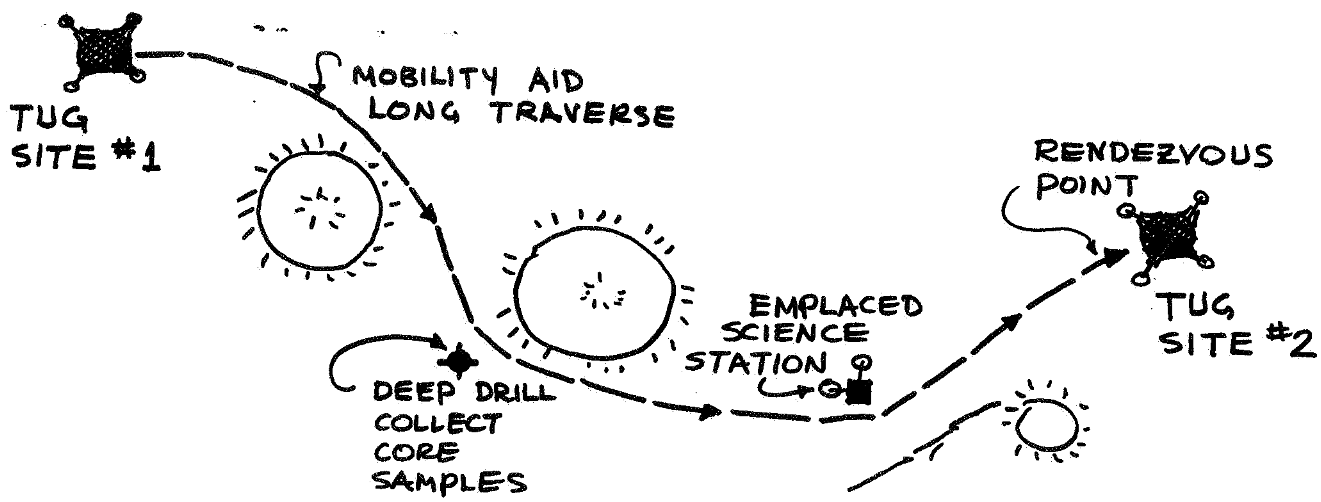


FIGURE 5.8-1. UNMANNED LONG TRAVERSE

<u>Mission 1</u> - Geology and Geophysics Experiments

Duration: 14 days to 3 months

Payload: Deep core drill and measuring devices

Weight = 1670 lbs., vol. = 35.3 cu. ft.

Distance: Around 600 km.

Experiment Time: 60 to 120 hours - vehicle stationary

Travel Time: 200 to 275 hours - vehicle moving at 2 to 3 km/hr

Activity: System must carry deep core drill and sample containers, must stop 1 or 2 times along the traverse to deploy drill and drill a 100 ft. hole. Must collect 20 samples in containers and deliver samples to a rendezvous point for return to Earth.

<u>Alternate</u> mission would be to emplace science stations but not drill holes. Assuming that less time would be taken in deploying stations, a range of 800 to 1000 km might be expected.

<u>Mission 2</u> - Geodesy Experiments

Duration: 14 days to 3 months

Payload: Selenodetic equipment, cameras, gravimeters, seismometers, laser retroreflector, neutron-gamma analyzer, spectrometers, magnetometer, electric field gauge, and solar wind spectrometer.

Weight = 500 lbs., vol = 20 cu. ft.

Distance: 1000 to 1500 km

Experiment Time: Continuous while vehicle is in transit. Probably would account for about 200 hours.

Travel Time: 600 hours - vehicle moving at 2 km/hr.

Activity: System continuously or intermittently collects and transmits data related to surface features and fields. Data is transmitted to LOSS or Earth.

<u>Mission 3</u> - Geodesy Experiments

Duration: 36 hours

Payload: Same as Mission 2 plus crew

Distance: Up to 250 km

Experiment Time: Continuous, about 20 hours

Travel Time: 20 hours at 10 km/hr, stop to sleep for 14 hours

Activity: Continuous collection and transmission of data.
Testing of crew functions.

5.8.2. Manned Medium Traverse

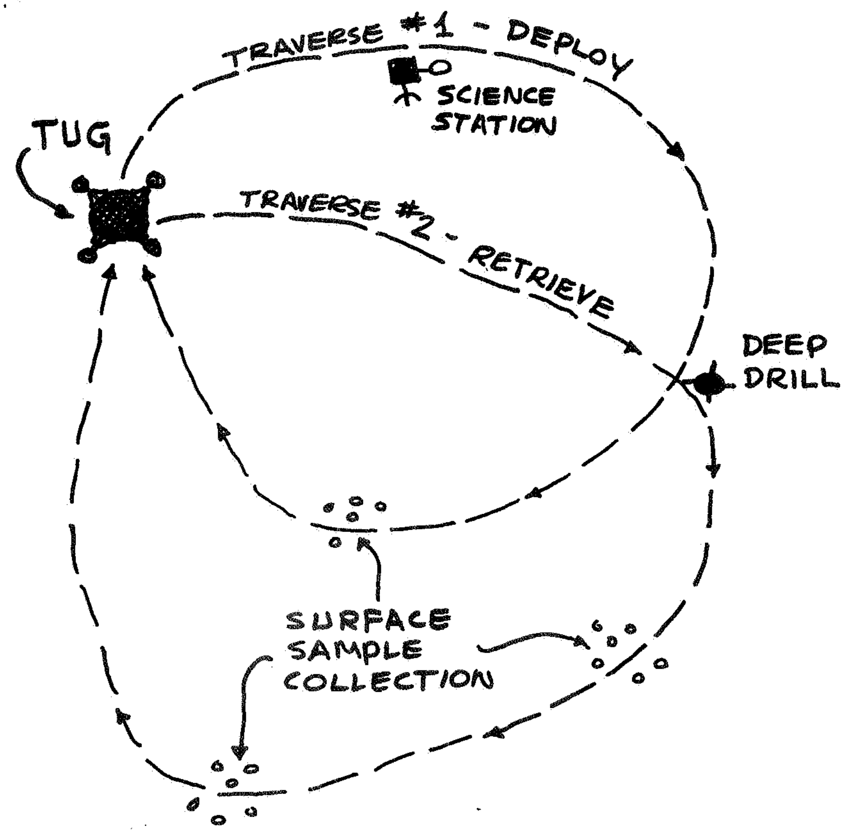


FIGURE 5.8-2. MANNED MEDIUM TRAVERSE

Mission 4 - Geology and Geophysics Experiments

Duration: 36 hours

Payload: Deep drill or science stations, core containers, soil mechanics device, geology tools.
Weight= 1870 lbs. plus crew, Volume = 35 cu. ft.

Distance: About 200 km

Experiment Time: 8 hours, vehicle stationary

Travel Time: 20 hours, sleep 14 hours

Activity: Deploy drill or science stations, collect surface samples with geology tools, retrieve drill and core samples. Surface samples to be collected will be up to about 150 lbs. in weight.

5.8.3. Manned Short Sortie

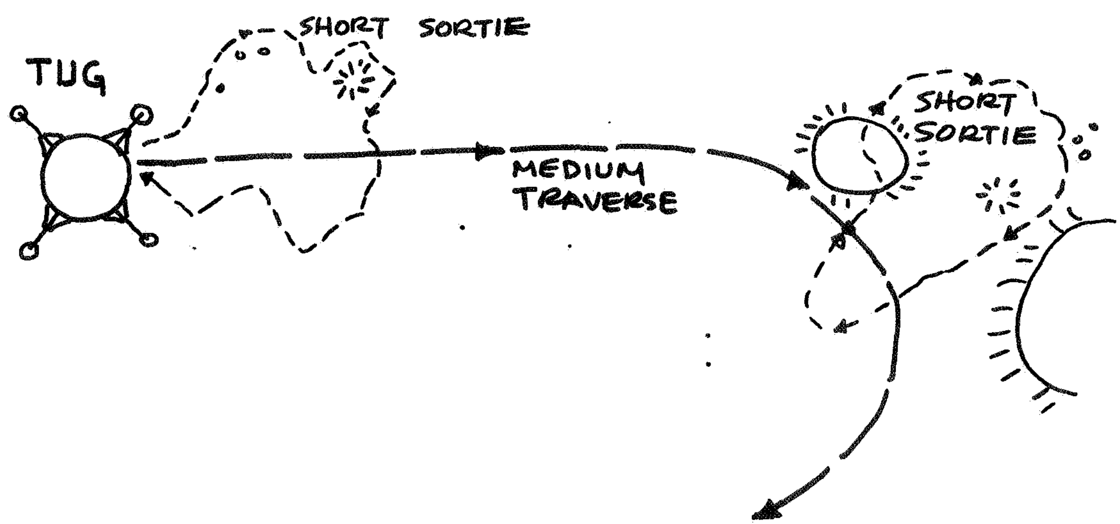


FIGURE 5.8-3. MANNED SHORT SORTIE

Mission 5 - Visual Inspection

Duration: 1 to 8 hours

Payload: Cameras, geology tools, portable mass spectrometer, some surface samples (about 80 lbs.). Payload Weight - 165 lbs. plus samples.

Distance: 5 km or less

Experiment
Time: 1 to 7 hours

Travel
Time: Up to 3 hours

Activity: Manned sortie from the Tug site or from the medium traverse for purposes of visual inspection of interesting lunar features and correlation with orbital mapping. Detection and inspection of possible sites for the LSB and locations of scientific opportunity. Could also retrieve drill samples and collect surface samples. Desirable to access hard-to-get locations -- such as, crater bottoms, narrow passages, crevasses, faults, etc.

5.8.4. Lunar Base Construction

Mission 6 - Manned Lunar Base Construction

Duration: 1 to 6 hours

Payload: None except construction aids

Distance: 10 km

Working
Time: 6 hours

Moving
Time: 6 hours

Activity: Leveling soil, cutting trenches, building ramp, welding

Mission 7 - Unmanned Lunar Base Construction

Duration: 14 days

Payload: Construction equipment

Distance:	300 km
Working Time:	100 hours
Moving Time:	170 hours plus 100 hours while working
Activity:	Traversing then leveling soil, cutting trenches.

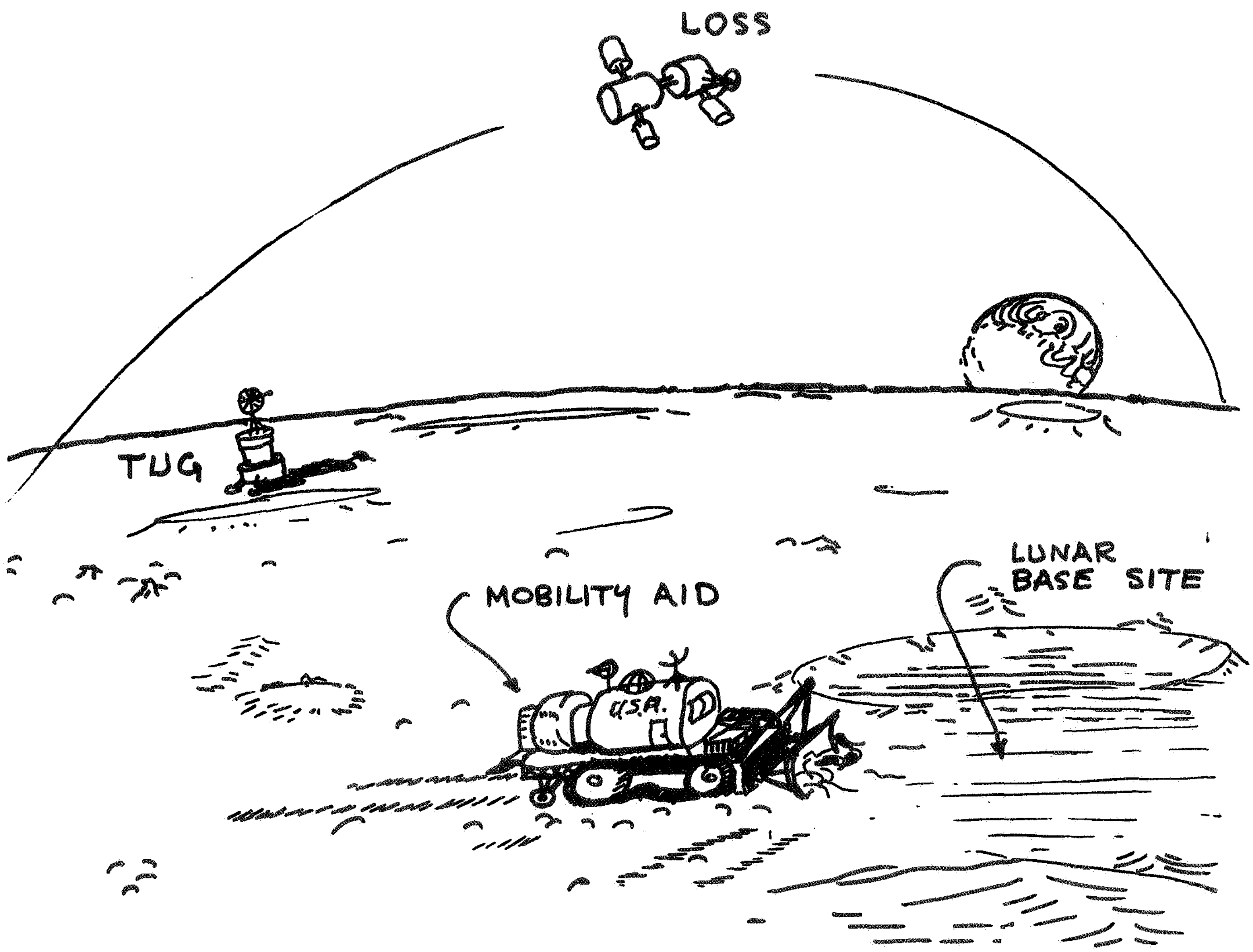


FIGURE 5.8-4. A CONSTRUCTION AID

5.8.5. Lunar Base Support

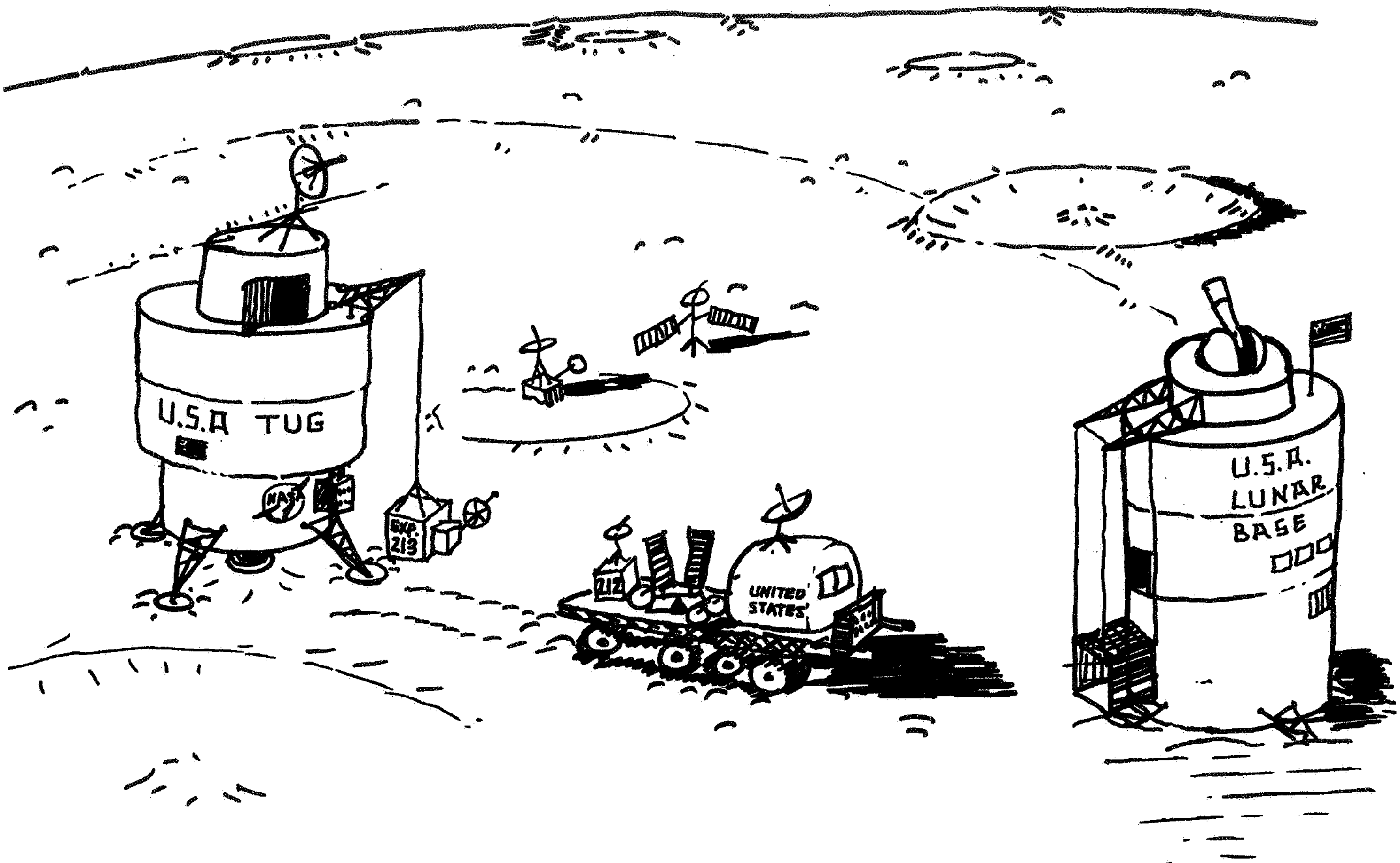


FIGURE 5.8-5. LUNAR BASE SUPPORT

<u>Mission 8</u> - Manned Lunar Base Support

Duration: 16 hours

Payload: 2000 lbs. of supplies and fuel for the base

Distance: 6 km

Working Time: 1 hour transporting plus 12 hours to load/unload

Moving
  Time: 1 hour

Activity: Supply Lunar Surface Base by transporting equipment from Tug to base and transporting lunar samples from base to Tug for return to Earth. Transport fuel from Tug to LSB storage bay and assists in attaching/ detaching fuel units. Transport devices and tools used in construction and maintenance of base.

<u>Mission 9</u> - Unmanned Lunar Base Support

Duration: 160 hours

Payload: 2000 lbs.

Distance: 6 km

Working
  Time: 6 hours transporting plus 12 hours to load/unload

Moving
  Time: 6 hours

Activity: Same as Mission 8

<u>Mission 10</u> - Manned personnel Transport

Duration: 6 hours maximum

Payload: 4 or 5 persons to Lunar Base and return same number to Tug. Weight up to 2000 lbs.

Distance: 6 km

Working
  Time: 1 hour in transit plus 4 hours entering/exiting

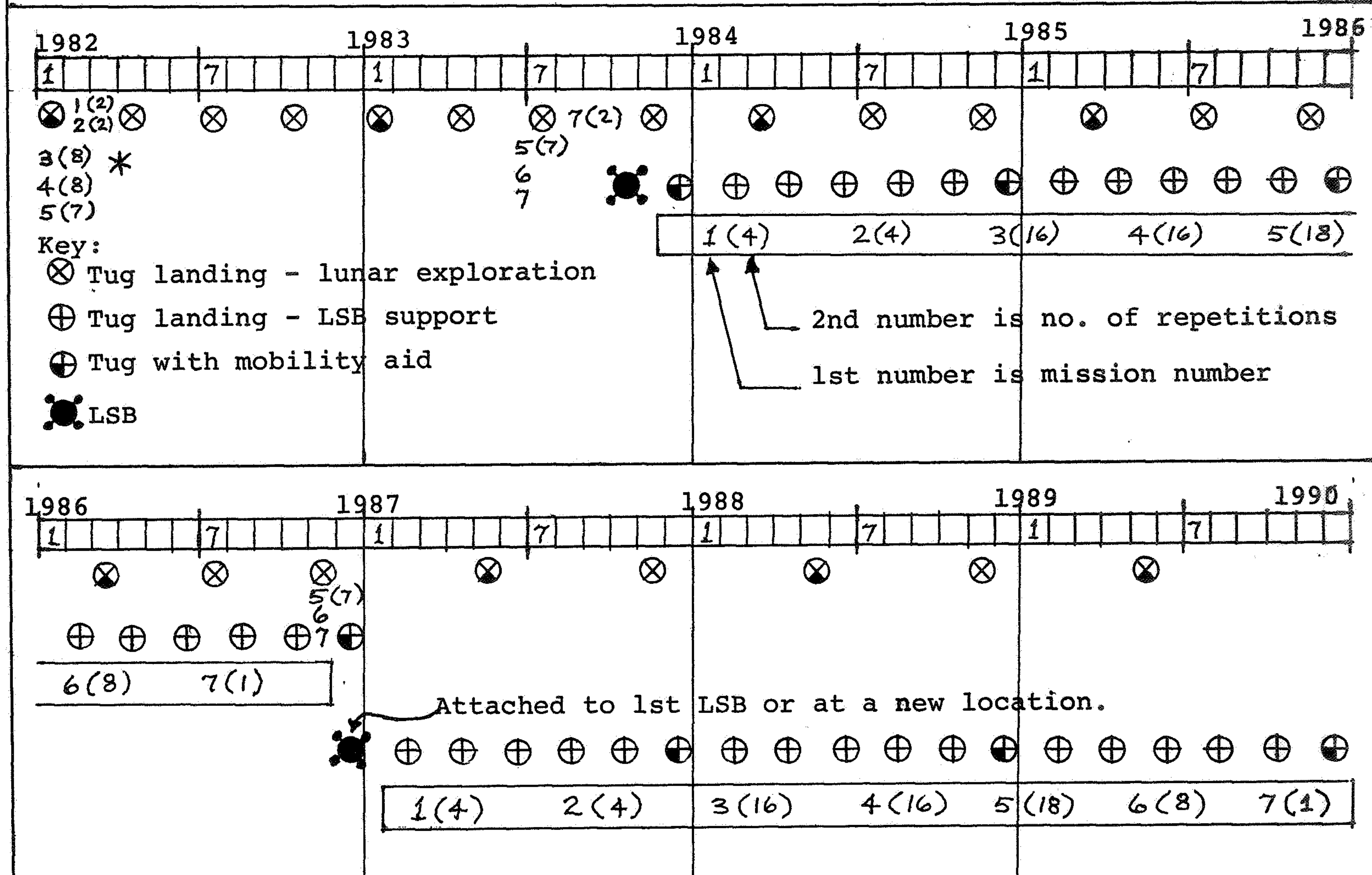
Moving
  Time: 1 hour

Activity: Transport LSB personnel between LSB and the Tug

5.9. Tug Landing Schedule

Based on the definition of the ten missions in the preceding section and the Integrated Program Plan a schedule of Tug landings and lunar missions was made in order to indicate the number of mobility aids which would be necessary in the 1980 period. From this schedule it was also possible to show the relative magnitude of the manned vs unmanned use of the mobility system and the total number of each mission. A summary of these items is shown following the schedule.

TABLE 5.9-1 SCHEDULE OF TUG LANDINGS AND LUNAR MISSIONS



* These missions are repeated at all other exploratory landings except where shown differently.

TABLE 5.9-2 SUMMARY OF TUG SCHEDULE

A. Mission Distribution:

Mission Number	Total Number	Manned	Unmanned
1	50		X
2	50		X
3	192	X	
4	192	X	
5	190	X	
6	18	X	
7	7		X
8	18	X	
9	19		X
10	18	X	
Total	754	628	126

B. Tug landing:	Lunar exploration	22
	LSB support	39
	Total	61

Mobility aids;	Lunar Exploration	8
	LSB support	7

REFERENCES

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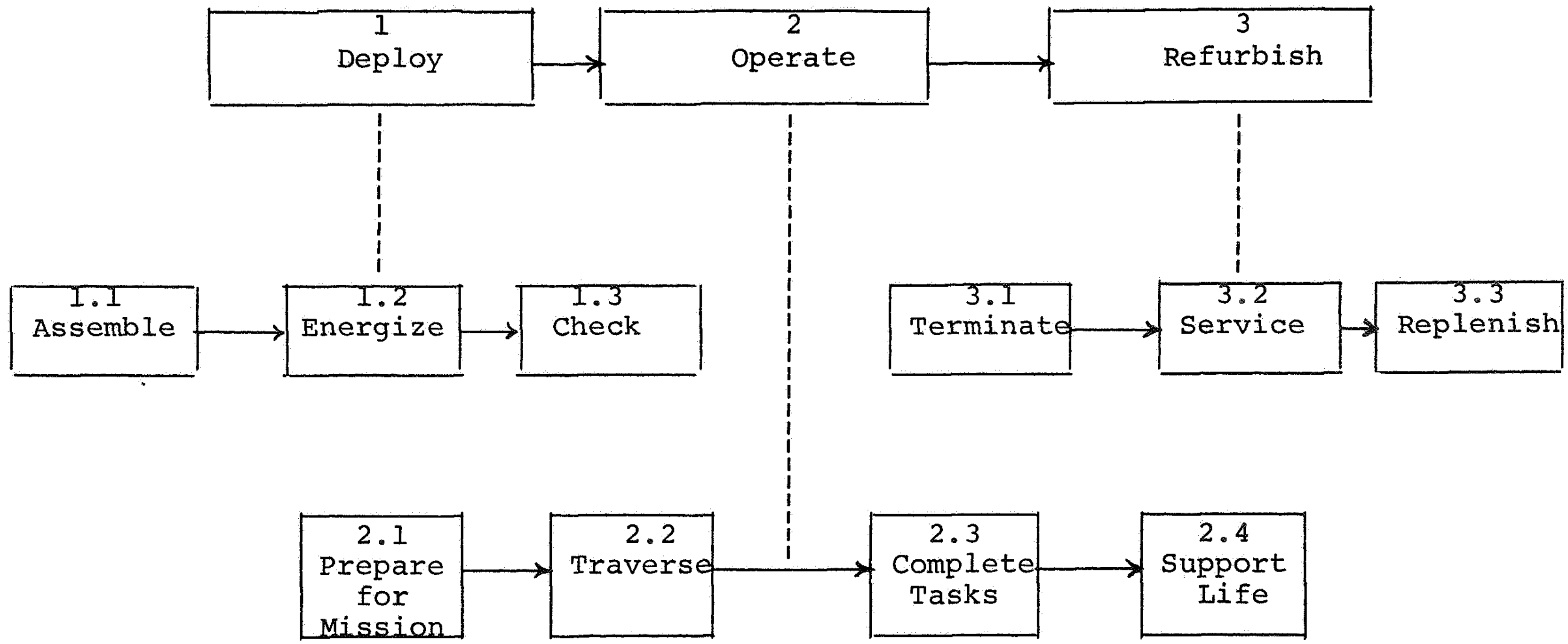
CHAPTER 6

FUNCTIONAL ANALYSIS

Raymond F. Neathery

A generally accepted first step in Systems Engineering is a Functional Analysis. The functional analysis is generated by determining what functions are required to meet the gross requirements of the system. The functional analysis should not be hardware oriented. It is generally generated by considering successively more specific levels. Functional requirements, and in turn, design requirements may be determined from the functional analysis. The documentation of the functional analysis insures that all functions and interfaces have been considered.

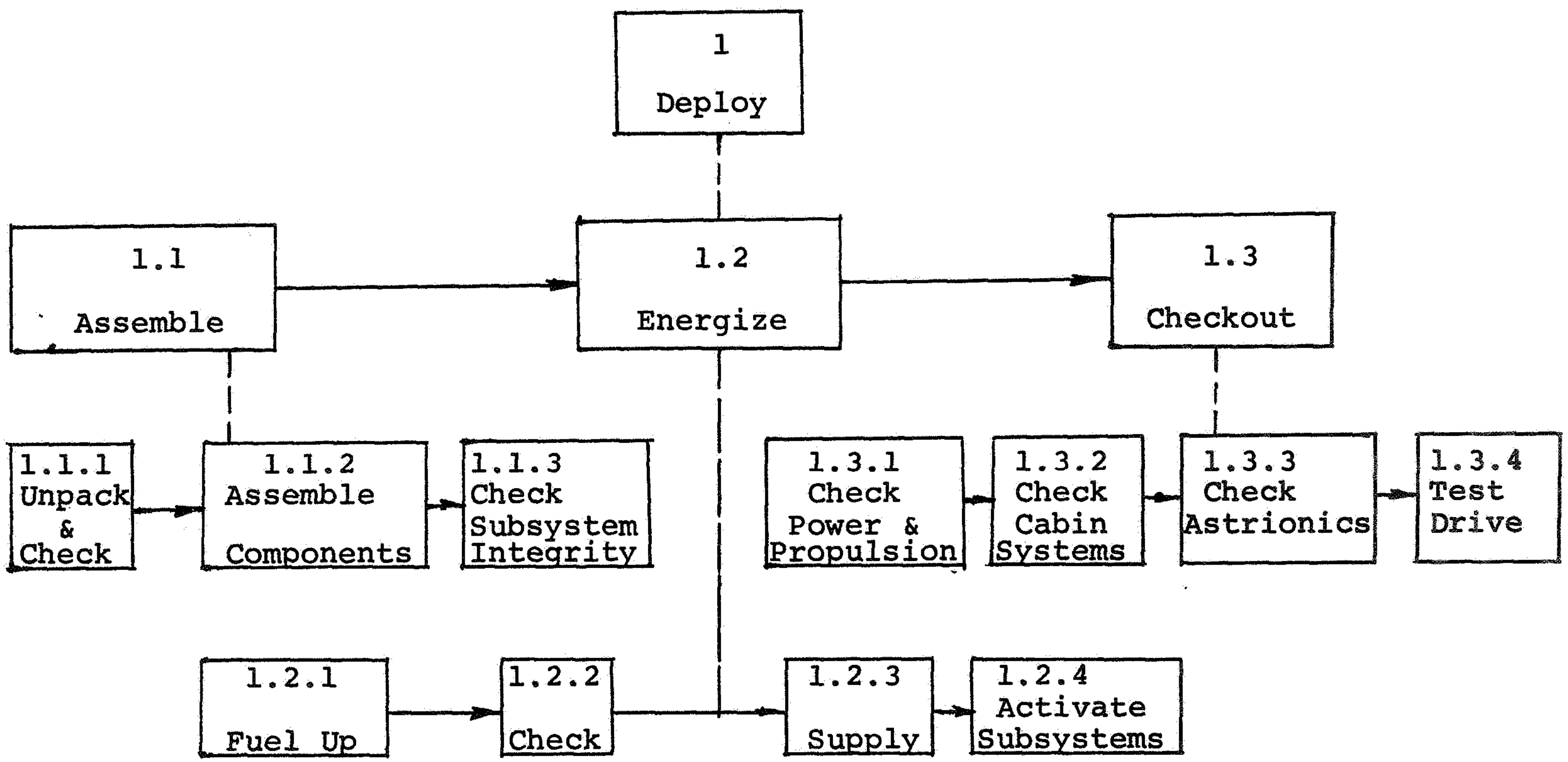
In this chapter a functional analysis to the third level and the accompanying functional requirements of the mobility system are given. Fourth level analyses were done for the missions and are included in Appendix D.



MOBILITY SYSTEM FUNCTIONAL ANALYSIS

FIRST & SECOND LEVEL

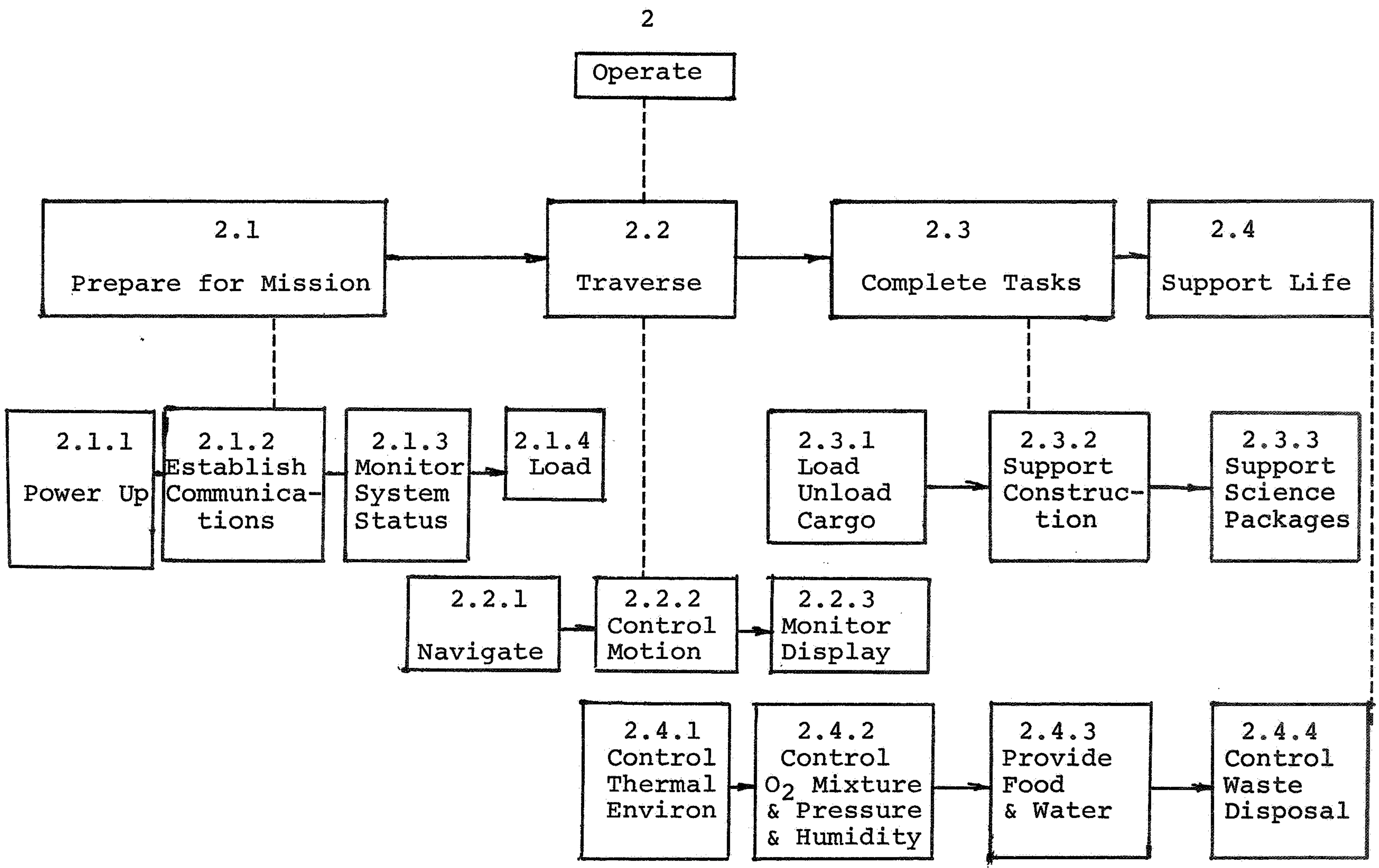
FIGURE 6-1



MOBILITY SYSTEM FUNCTIONAL ANALYSIS

THIRD LEVEL (DEPLOY)

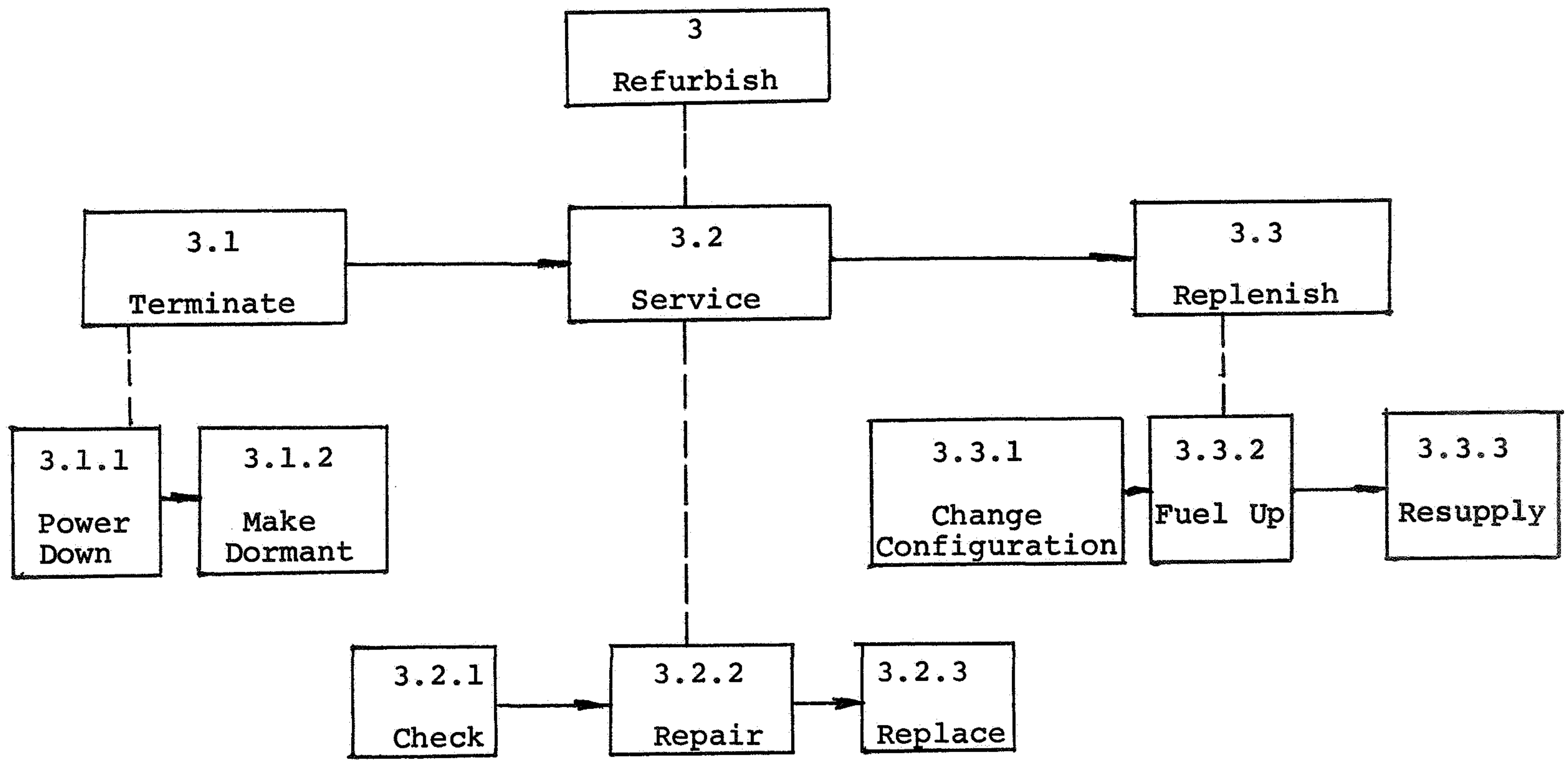
FIGURE 6-2



MOBILITY SYSTEM FUNCTIONAL ANALYSIS

THIRD LEVEL (OPERATE)

FIGURE 6-3



MOBILITY SYSTEM FUNCTIONAL ANALYSIS

THIRD LEVEL (REFURBISH)

FIGURE 6-4

TABLE 6-1

THIRD LEVEL FUNCTIONAL REQUIREMENTS

FUNCTION	FUNCTIONAL REQUIREMENT
1.1.1 Unpack and Check	Structure must be freed for assembly and integrity and completeness must be checked.
1.1.2 Assemble Components	The system must be structurally ready for operation.
1.1.3 Check Subsystem Integrity	It must be verified that the system is structurally ready for operation.
1.2.1 Fuel Up	Sufficient energy must be supplied to the system for successful completion of mission and to allow for contingencies.
1.2.2 Check	It must be verified that fuel levels are adequate, operable and non-hazardous.
1.2.3 Supply (for check-out)	Non-fuel consumables necessary for operation must be loaded.
1.2.4 Activate Subsystems	All subsystems requiring power must be put in the operational or stand-by mode.
1.3.1. Check Power and Propulsion	It must be verified that all power and propulsion are GO.

1.3.2	Check Cabin Systems	It must be verified that all cabin systems are GO.
1.3.3	Check Astrionics	It must be verified that all Astrionic Systems are GO.
1.3.4	Test Drive	The system must be operated and it must be determined if all subsystems are functional. The system must be returned to the stand-by mode.
2.1.1	Power Up	Power must be supplied to all subsystems for mission operation.
2.1.2	Establish Communications	Communications access must be provided between all stations involved.
2.1.3	Monitor System Status	It must be verified that all subsystems are GO.
2.1.4	Load (for traverse)	The system must be loaded with all necessary personnel, cargo and equipment.
2.2.1	Navigate	The location of the vehicle must be determined with sufficient accuracy to accomplish mission and provide rescue capability.
2.2.2	Control Motion	The motion of the system must be initiated, controlled and terminated in both manned and unmanned modes.

2.2.3	Monitor Display	All systems must be monitored at appropriate intervals and necessary corrective action must be taken.
2.3.1	Load and Unload Cargo	Cargo and crew must be loaded and unloaded in a lunar environment without adversely affecting their intended use or life.
2.3.2	Support Construction	The system must support lunar base construction.
2.3.3	Support Science	The system must supply mechanical and electrical power assists when required of scientific or technical missions.
2.4.1	Control Thermal Environment	The correct operating temperature range for all systems must be maintained and verified.
2.4.2	Control O_2 Mixture Pressure & Humidity	A 70% O_2-30% N_2 mixture must be maintained at 3.5 to 5.0 psi and 30 to 70% relative humidity.
2.4.3	Provide Food and Water	Appropriate food and water and and mode of consumption must be provided on manned missions.
2.4.4	Control Waste Disposal	Provision must be made for the collection and storage of urine, feces, waste food, wrappers, etc.

3.1.1	Power Down	All live subsystems must be deactivated (communications must not be destroyed) and fuel systems made dormant.
3.1.2	Make Dormant	Equipment must be protected and stored while allowing for maintenance access.
3.2.1	Check	All subsystems must be checked for wear and damage. Necessary repair and replacement must be identified. Items which cannot be repaired or replaced must be identified and reported.
3.2.2	Repair	Necessary repairs must be made, operation verified and reported.
3.2.3	Replace	Necessary replacements must be made, operation verified and reported. Light consumables must be added.
3.3.1	Change Configuration	The configuration must be altered to meet the requirements of the next mission.
3.3.2	Fuel-Up	Same as 1.2.1
3.3.3	Resupply	Same as 1.2.3.

PART II

METHODS

In this part of the report the methods used to generate a system concept will be discussed. The discussion begins with a presentation of the various candidates that were considered. This followed by a discussion of two evaluation procedures and a method of synthesis. Appendix F discusses the philosophy of these methods.

CHAPTER 7

CANDIDATE SYSTEMS

Raymond F. Neathery

In Phase I of this program much effort was devoted to generating mobility system concepts. Practically all of the team members participated in this process. This chapter documents the results of that effort.

7.1 General Classes

There are four classes of vehicles which merit serious consideration. The Rover and Flyer classes are obvious candidates and several of these have been proposed to NASA[1]. Two less obvious candidates are the Ground Effects Machine (GEM) and the Hopper. NASA has received a proposal on the GEM[2] and a serious study has been made of the Hopper[3].

All of the vehicles discussed here are of a scale consistent with the gross system requirements.

The Rover is any vehicle which depends on physical contact with the surface for suspension such as wheeled or tracked vehicles. Flyers are vehicles that depend on a rocket thrust for support. A GEM depends on the pressure of a contained gas. A Hopper depends primarily on a ballistic trajectory for suspension when in motion. Further, it mechanically conserves the energy of impact. These general classes of vehicles are shown schematically in Figure 7.1-1.

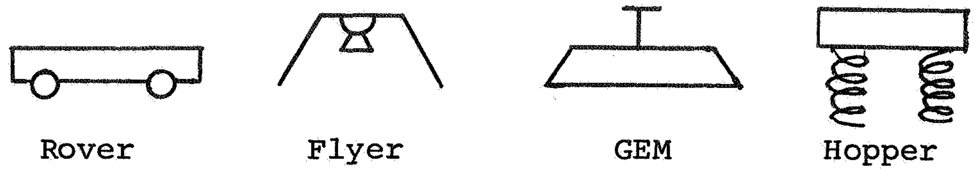


FIGURE 7.1-1 GENERAL CLASSES OF VEHICLES

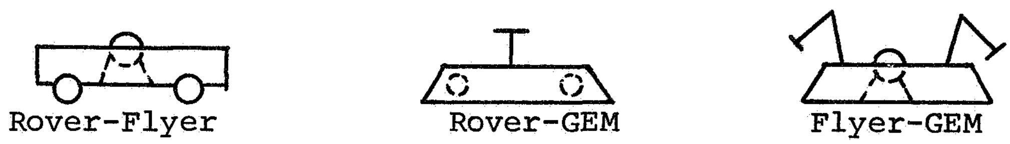


FIGURE 7.2-1 HYBRID VEHICLES

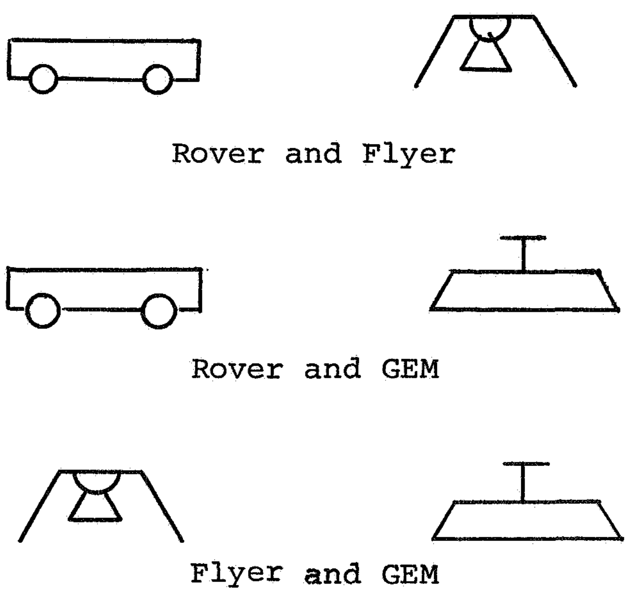


FIGURE 7.2-2 PARALLEL COMBINATIONS OF VEHICLES

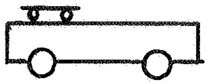
7.2 Hybrids and Combinations

A hybrid is defined here as a vehicle which has the characteristics of both component vehicles. A combination is a system which consists of two separate vehicles. Two types of combinations were generated. Parallel combinations are combinations where each vehicle is of the same order of magnitude and approximately meets the gross system requirements. Piggy-back combinations are combinations where the carrier vehicle approximately meets the gross system requirements and the second vehicle is small enough to be carried by the carrier.

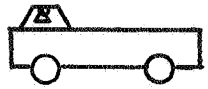
The Hopper seemed to hold little promise and, therefore, was not permuted with other vehicles in generating hybrids and combinations.

There are three possible hybrids. These are shown in Figure 7.2-1. Of these, a Flyer-GEM offers two compatible components without any obvious severe penalties.

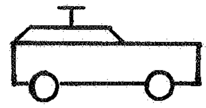
There are nine possible piggy-back combinations as shown in Figure 7.2-2. The Rover as a carried vehicle is a particularly strong candidate since the Lunar Rover[4] will have been developed and used. An upgraded version might be particularly attractive. The Rover would also be attractive as the carrier since it would allow a generous payload and would have a minimum of development risk.



Rover on Rover

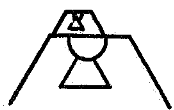


Flyer on Rover

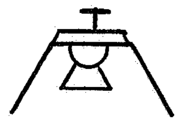


GEM on Rover

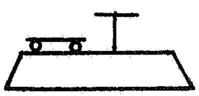
Rover on Flyer



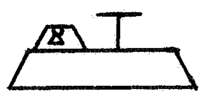
Flyer on Flyer



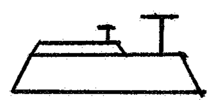
GEM on Flyer



Rover on GEM



Flyer on GEM



GEM on GEM

FIGURE 7.2-3 PIGGY-BACK COMBINATIONS OF VEHICLES

7.3 Novel Concepts

In this section several mobility concepts are presented, some of which involve unusually high development risks. However, in a rapidly advancing technology the development risk for some could rapidly change. Others, however, will simply remain as wild ideas. Also presented here are ideas which seemed promising but could not be adequately studied in the program.

A mobility system consisting of a glass tube generated from lunar soil using solar energy was proposed by Trieschmann. This system is described in Appendix C.

Several unusual modifications were proposed for Rover type vehicles. The following are illustrated in Figure 7.3-1.

1. Ski tracks (Snowmobile)
2. Rocket propulsion
3. Rocket propulsion using moon dust as the exhausted material (Dust Jet)
4. Rolling Sphere
5. Rolling Cylinder
6. Crawler

Other mobility systems are pictured in Figure 7.3-2. The Biped and Quadruped could be programmed or slaved to a man on board, or remotely. The Ballistic Deliver could be a cannon type or carry its own propellant. Packaging is, of course, the major problem. The Porcupine Ball consists of fully retractable struts around a sphere. The Sky and Hopper is a variation of the Flyer where thrusters are fired for short periods producing the illustrated trajectory. A similar surface bound device is a bouncing ball. Regardless of the mode of propulsion, any mobility system could benefit from a modular concept. A modular concept for a wheeled Rover is depicted in Figure 7.3-3. The basic element consists of an axle assembly including electric drive and braking in each wheel, and an unbilical connector for power and control. Additional elements are listed below:

1. Connecting Cables
2. Structural Connecting Bars
3. Astrionics Package
4. Battery Power Pack
5. Crew Station (with or without cab)
6. Trailer Bed
7. Life Support Trailer
8. Power Pack Station (batteries, fuel cell, RTG)
9. Construction Attachments

Possbile configurations of a modular system are illustrated. A few of the advantages of this system are as follows. With Lunar gravity, assembly will require little effort; commonality is maximized; additional tractive power is easily obtained by grouping drive wheels as illustrated in the construction configuration; a single astrionics package would be sufficient for a train; etc.

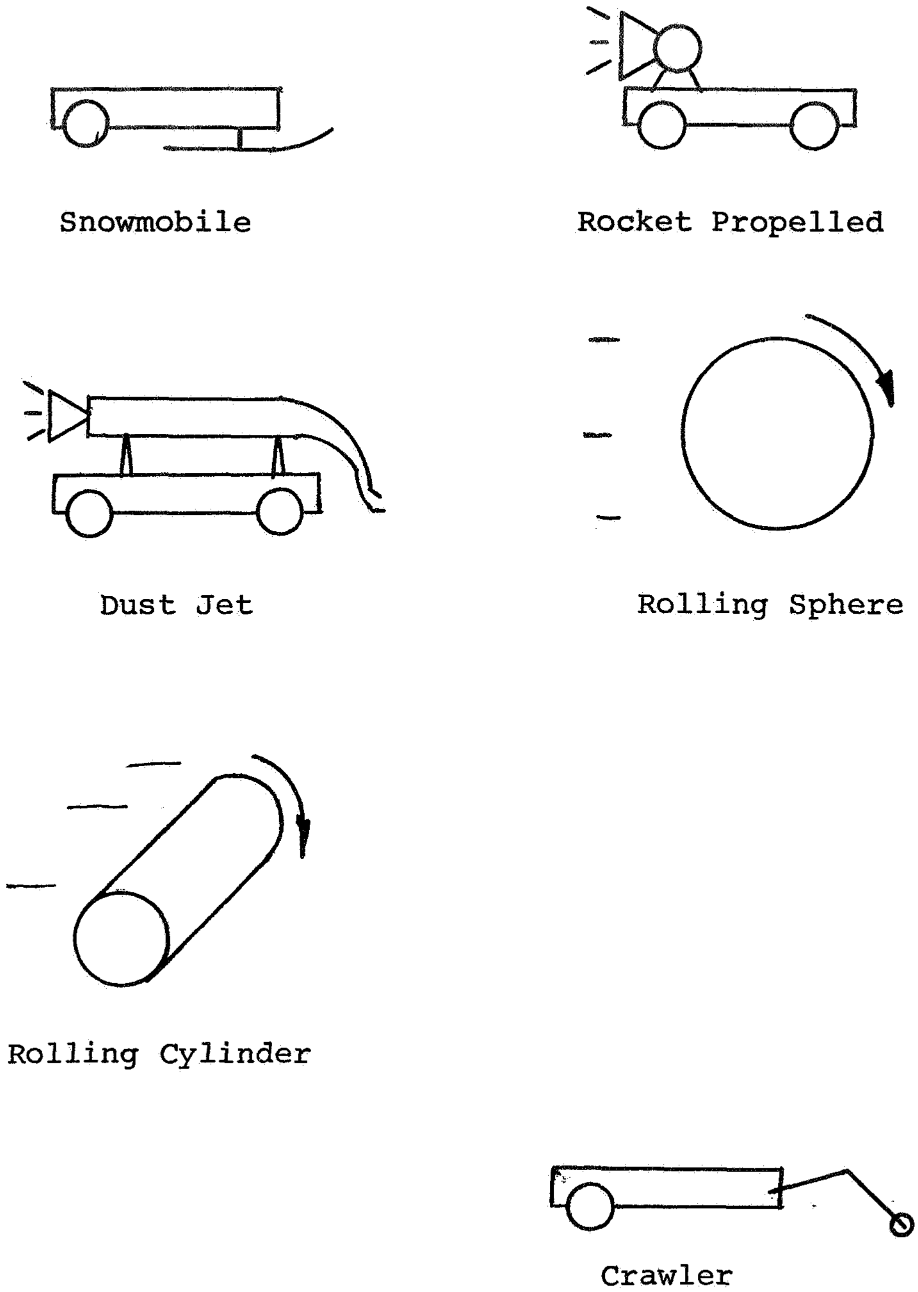


FIGURE 7.3-1 VARIATIONS OF THE ROVER

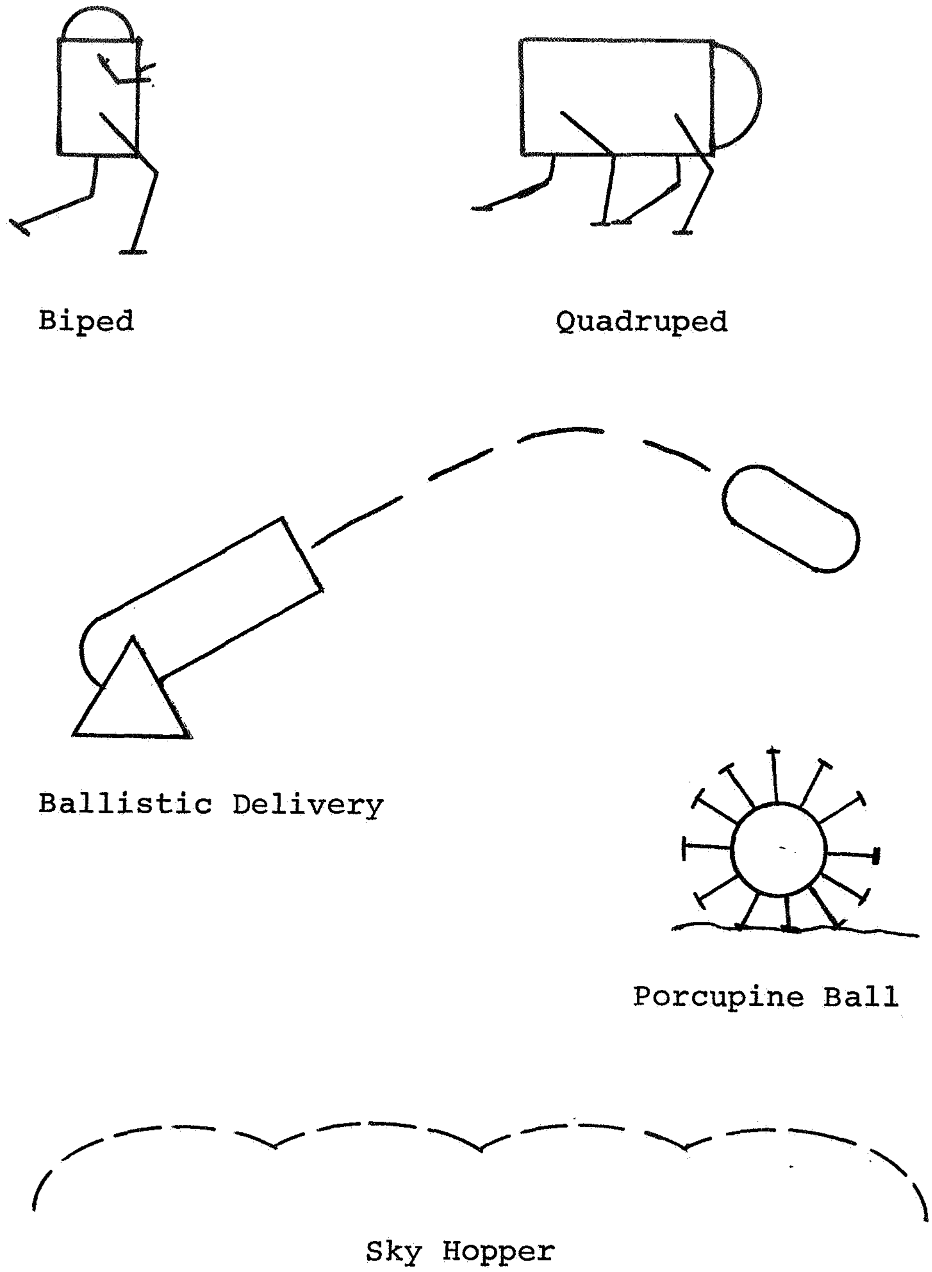


Figure 7.3-2 MOBILITY CONCEPTS

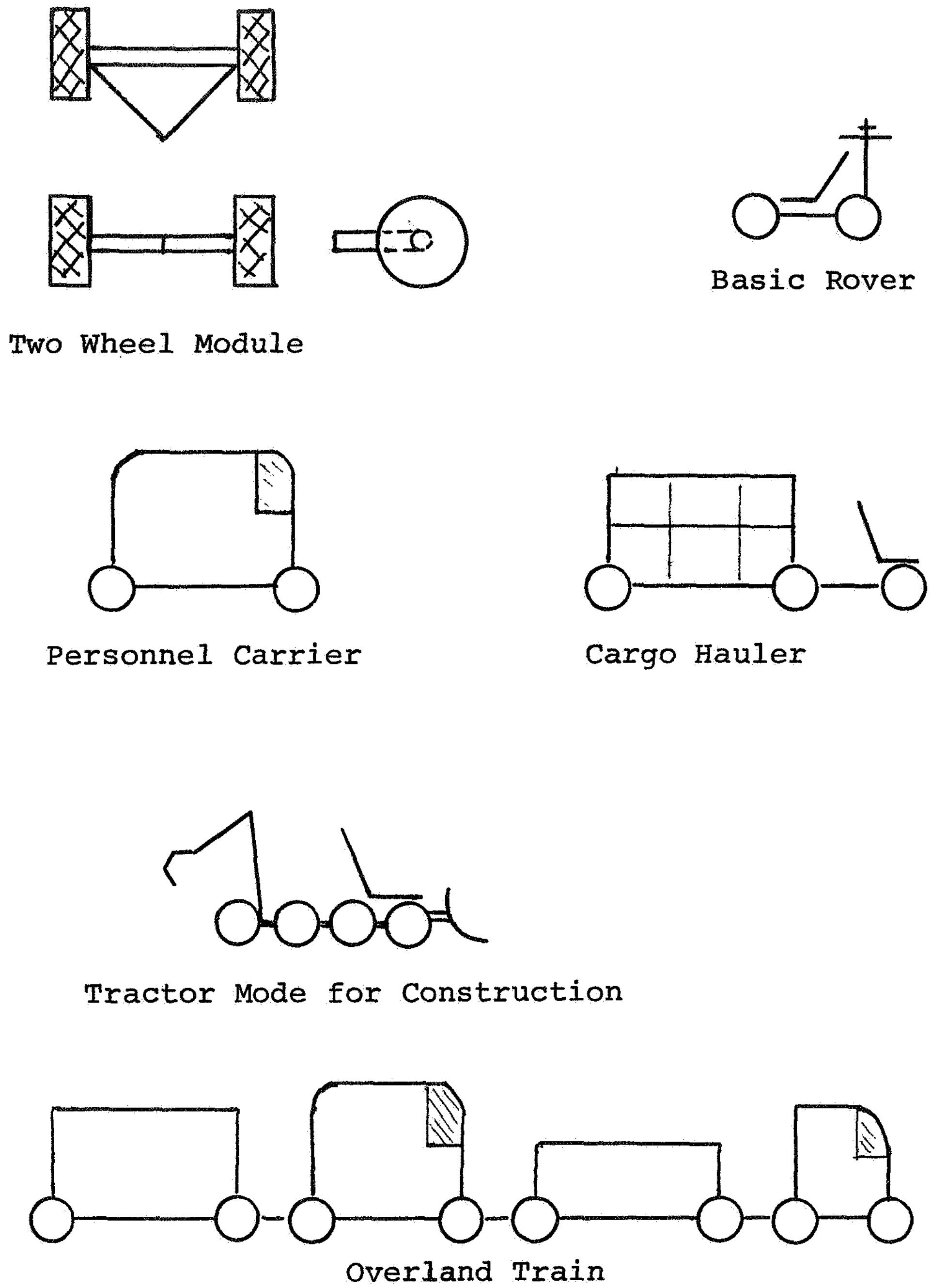


FIGURE 7.3-3 MODULAR CONCEPT

REFERENCES

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2. Unsolicited Proposal, "Lunar Ground Effects Machine", TRW Systems, October, 1969.

3. Kaplan, M. H. and Seifert, H. S., "Hopping Transporters for Lunar Exploration", Journal of Spacecraft and Rockets, Vol. 6, No. 8, August, 1969, pp.917-922.

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CHAPTER 8

EVALUMATRIX

J. C. Lindholm

In the early part of Phase I brainstorming sessions were held at group and team levels. The purpose of these sessions was to develop an extensive list of possible mobility candidates. Once such a list was started it was evident that a systematic method or methods of evaluating them was needed. Two methods of evaluation were developed. The one to be discussed in this chapter is the Evalumatrix. An evalumatrix is a matrix array with the proposed candidates as the column axis and the evaluating criteria as the other axis. When the array has been completed the columns can be added to get the evaluation. To generate the matrix it was necessary to establish criteria for rating the various proposed candidate systems. A set of parameters was developed as evaluating criteria. These parameters were considered important for the safe operation of the mobility system and successful completion of mission tasks.

Each of the four groups developed its own set of parameters which were considered of prime importance to its activity. Some overlap of parameters selected resulted. The following sections give in detail the development procedure used by each group to generate parameters and evaluate the candidate systems. In some cases the parameters were weighted as to importance in accomplishing the desired output. Each candidate was then rated as to how well it satisfied each of the weighted parameters. The totals for each candidate were obtained which resulted in a ranking of candidates according to how well each met the parametric requirements.

A total of thirteen candidates for the mobility system were given serious consideration. These are described in Chapter 7 and can be considered as four basic types, or combinations of these basic types. The rover, GEM (ground effects machine), flyer and hopper were considered the basic types, i.e. primary candidates. Some of the groups evaluated only the primary candidates while others also considered the secondary candidates.

The results of each evalumatrix was normalized with respect to the top candidate which was assigned a value of 100. Each set of results was weighted equally in determining the composite evaluation of the candidate mobility systems. Table 8-1 gives the results for the four basic types of systems. As can be seen from the table the rover vehicle was considered best in all four evaluations. The GEM was considered second and worthy of further consideration. Of the secondary candidates, the ones that incorporated the rover as part of the system ranked well as possible systems. These are discussed in Sections 8.2 and 8.4.

TABLE 8-1. EVALUMATRIX FOR FOUR PRIMARY CANDIDATES

Candidate Group	Rover	GEM	Flyer	Hopper
Power and Propulsion	100	82	45	45
Astrionics	100	98	78	58
Human Factors	100	83	72	42
Configuration	100	76	60	43
Average	100	85	66	47

8.1. Power and Propulsion Performance Matrix

S. J. Clark

In a systems evaluation, vehicle performance should be rated in terms of operational and design parameters. This requires the establishment of either dimensionless indices or composite dimensionless factors. The indices that can be used are numerous. It is, therefore, important that the indices which are selected are the ones that are meaningful in terms of the mission requirements of the vehicle.

The initial approach was to set up two performance matrices, one for sources of electrical power and another for propulsion systems. These MxN matrices were as follows:

M. Power Mode

1. Nuclear reactors
2. Radioisotope thermoelectric generators
3. Fuel cells
4. Solar panels
5. Batteries
6. Reaction jets
7. New concepts

N. Performance Constraint Indices

1. Power per unit mass
2. Power per unit volume
3. Efficiency
4. Peak power capability
5. Power response
6. Refueling requirements
7. Simplicity
8. Temperature operating limits
9. Orientation requirements
10. Shielding requirements
11. Control requirements

M. Propulsion Mode

1. Wheels
2. Tracks
3. Ground effects machine
4. Flyers
5. Hybrids
6. New concepts

N. Performance Constraint Indices

1. Pull capability
2. Moving resistance coefficient
3. Power efficiency
4. Flotation coefficient
5. Crevice and ditch crossing
6. Step and obstacle crossing ability
7. Ride quality
8. Braking ability
9. Steering ability
10. Maintainability
11. Compatibility with power modes

Before significant work was done on the collecting of information for these matrices, there was an immediate need for fuel consumption, speed, and range data for specific classes of vehicles. To provide this information within the time allotted, a new approach was followed; a composite performance matrix was developed for combinations of power and propulsion systems. Performance parameters were developed which allowed direct comparisons between rovers, ground effects machines, flyers, and hoppers. These parameters were as follows:

1. $\dfrac{\text{Payload}}{\text{Gross Weight}}$

where

Gross Weight = Vehicle + Fuel + Payload (Weights)

2. $\dfrac{\text{Fuel Consumption Non-Stop Operation}}{\text{Fuel Consumption Start/Stop Operation}}$

3. $\dfrac{\text{Kilometers per pound of fuel}}{\text{Kilometers per pound of fuel (Best Vehicle)}}$

4. $\dfrac{\text{Speed, Kilometers/hr}}{\text{Speed, Kilometers/hr (Best Vehicle)}}$

Parametric curves were plotted for ground effects machine, flyers and rovers. These are included in Sections 8.1.1, 8.1.2 and 8.1.3.

Calculations were made for two loaded vehicle weights (fuel weight was not included since it would vary tremendously between vehicles). The loaded weights selected were 1500 and 7800 pounds. The methods for obtaining the graphical data are outlined in Sections 8.1.1, 8.1.2 and 8.1.3. The values for the hopper were obtained from published material.

The final performance matrix is shown in Table 8.1-1. Note that two weighting factors were used in arriving at the final figures. A mission weighting factor was used to account for the fact that the coefficients have more or less importance for certain types of missions. The ten missions outlined in Chapter 5 are basically four types of missions from the standpoint of power and propulsion. These are:

1. The long unmanned mission (1500 kilometers)
2. The 36 hour, 250 kilometer manned mission
3. The short sorties
4. Base support and supply

TABLE 8.1-1. VEHICLE CLASS PERFORMANCE MATRIX FOR POWER AND PROPULSION

Parameter	Wt Factor	Mission	Mission Wt Factor	Flyer	GEM	Rover	Hopper
Payload / Gross Weight	40	1	0.20	0.0	1.6	5.6	0.32
		2	0.45	3.6	14.4	14.4	0.72
		3	0.20	3.2	6.4	6.4	0.32
		4	0.15	0.0	4.8	4.8	0.32
Fuel NonStop / Fuel NonStop	20	1	0.20	2.4	3.6	4.0	4.0
		2	0.45	5.4	8.1	9.0	9.0
		3	0.20	2.4	3.6	4.0	4.0
		4	0.15	1.8	2.7	3.0	3.0
Kilometers / # Fuel	20	1	0.20	0.0	2.4	4.0	2.0
		2	0.45	0.2	5.4	9.0	4.5
		3	0.20	0.2	2.4	4.0	2.0
		4	0.15	0.1	1.8	3.0	1.5
Speed Ratio	20	1	0.20	4.0	0.2	0.0	0.2
		2	0.45	9.0	0.4	0.0	0.4
		3	0.20	0.0	0.0	0.0	0.0
		4	0.15	0.0	0.0	0.0	0.0
TOTAL SCORE				32	58	71	38
TOTALS BY MISSION		1		6.4	7.8	13.6	6.5
		2		18.2	28.3	32.4	14.6
		3		5.8	12.4	14.4	6.3
		4		1.9	9.3	10.8	4.7

The mission weighting factors were estimates based on the scientific importance of the missions. A second weighting factor was included to allow for the fact that the parameters are not of equal importance.

The point totals for the flyer, ground effects machine, rover and hopper were 32, 58, 71 and 32 respectively. These figures indicated that the flyer and hopper should be eliminated from further consideration. The flyer rated poorly due to high fuel consumption and a low payload to gross weight ratio for the longer missions. The hopper rated low because it had a very low payload to gross weight ratio.

It was felt that the ground effects machine should not be eliminated at this time since it did rate fairly well compared with the rover. There were also several factors that had not been included in the evaluation. Vehicular performance in regards to obstacle avoidance, operator comfort, and vehicle stability could improve the standing of the ground effects machine. Obstacles that could immobilize the rover or cause large amplitudes of displacement for a rover would pass under the ground effects machine without disturbing the vehicle at all (assuming a two-foot ground clearance).

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8.1.1. Analysis of Ground Effects Machine (GEM):

J. M. Ulrich

The study of a lunar GEM was necessarily based on design data reported by TRW[1]. Scaling factors developed in that report were used to study the effects of varying several parameters. The most important of these were plenum pressure, propellent flow rate, and vehicle weight. The critical design parameters which were of interest for comparison to the other classes of vehicles were (1) payload/gross weight vs range, (2) fuel consumption for steady operation, and (3) fuel consumption during starts and stops.

The first of the above parameters, i.e. payload/gross weight versus range was calculated based upon the following assumptions:

(a) constant vehicle velocity i.e. no accelerations or start-stop operation

(b) steering, drag and braking forces are neglected

(c) the constantly decreasing fuel load can be averaged for preliminary work

(d) soil absorption consumes approximately 50% of the plenum gas

In addition the TRW LUNAGEM baseline data used for scaling purposes were as follows:

DIAMETER	20 ft.
PLENUM AP	.005 psi
PROPELLENT FLOW	.015 lb/sec
SKIRT GAP	.5 in.

Calcuations were then made for varying payloads and gross weights, the latter being 1500 lb., 4500 lb., and 7800 lb. The results are shown in Figures 8.1-1 and 8.1-2 for the 1500 lb. and 7800 lb. gross weights, respectively.

The next set of curves, fuel consumption during steady operation vs. range, was generated from the data developed in step one:

$$\dot{W}_f \propto 2(\Delta P)^{\frac{1}{2}} \propto 2 \ (Wg)^{\frac{1}{2}} \qquad (8.1\text{-}1)$$

where $\dot{W}_f$ = reactant gas flow, lb/sec

Wg = gross vehicle weight, lb.

$$R = \frac{V \ W_f}{3600 \ \dot{W}_f} \qquad (8.1\text{-}2)$$

where R = range, KM

V = velocity of vehicle, KM/HR

W_f = total fuel load, lb

The results of these calculations are shown in Figures 8.1-3 and 8.1-4. The third set of calculations made were for fuel consumed during start-stop operation. It was assumed that braking would be done by some external means, so that only the energy required to accelerate the vehicle back up to a given speed need be considered. The reactants used in the calculations were:

(a) O_2/H_2 $\quad I_{sp} = 444$

(b) N_2O_4/N_2H_4-UDMH $\quad I_{sp} = 290$

The analysis was made as follows: Consider the fuel required to start from rest and accelerate the vehicle, using rocket propellants.

Payload To Gross Weight

Ratio Vs. Range

(For 1500 lb. Gross Wt.)

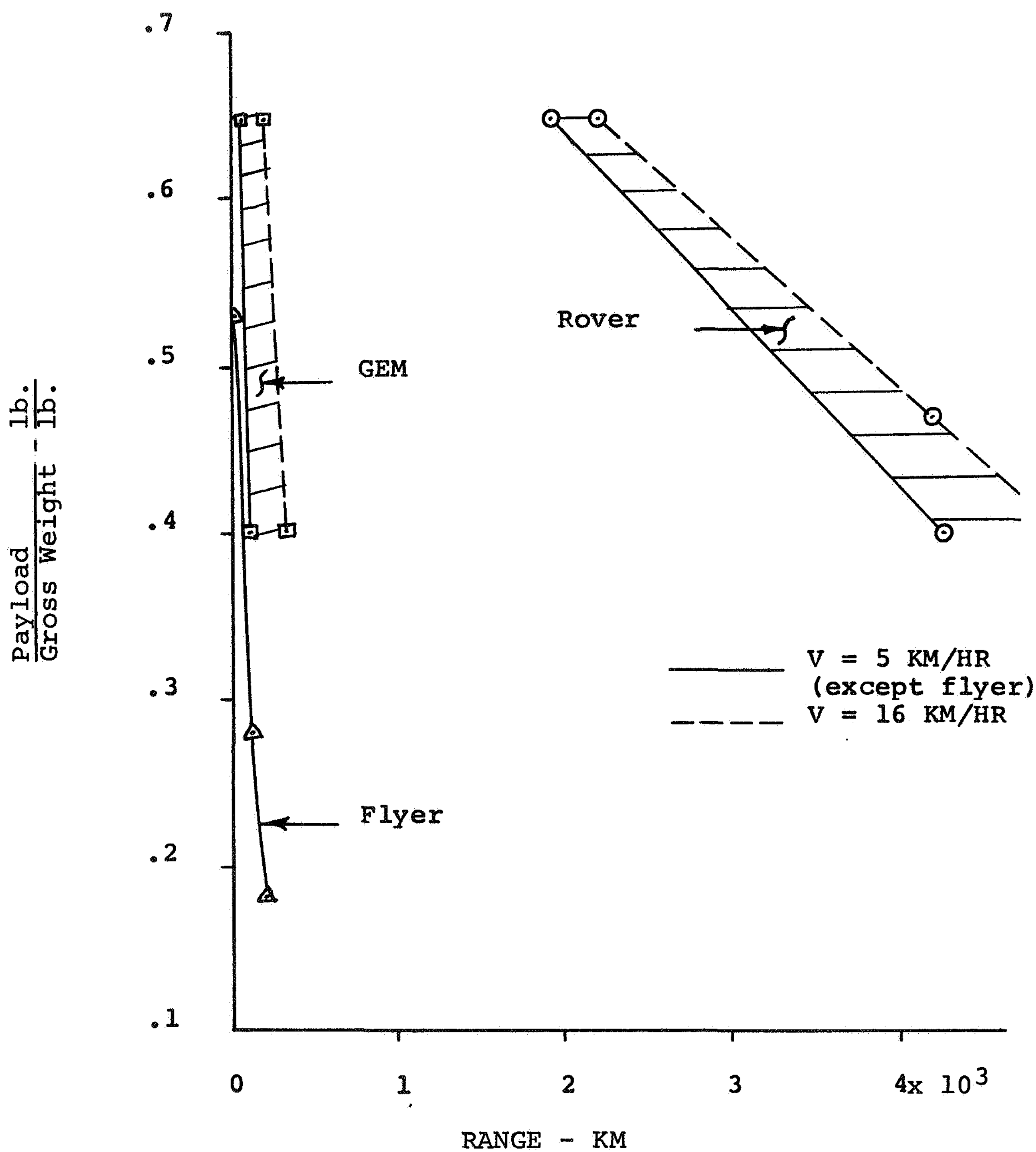


FIGURE 8.1-1

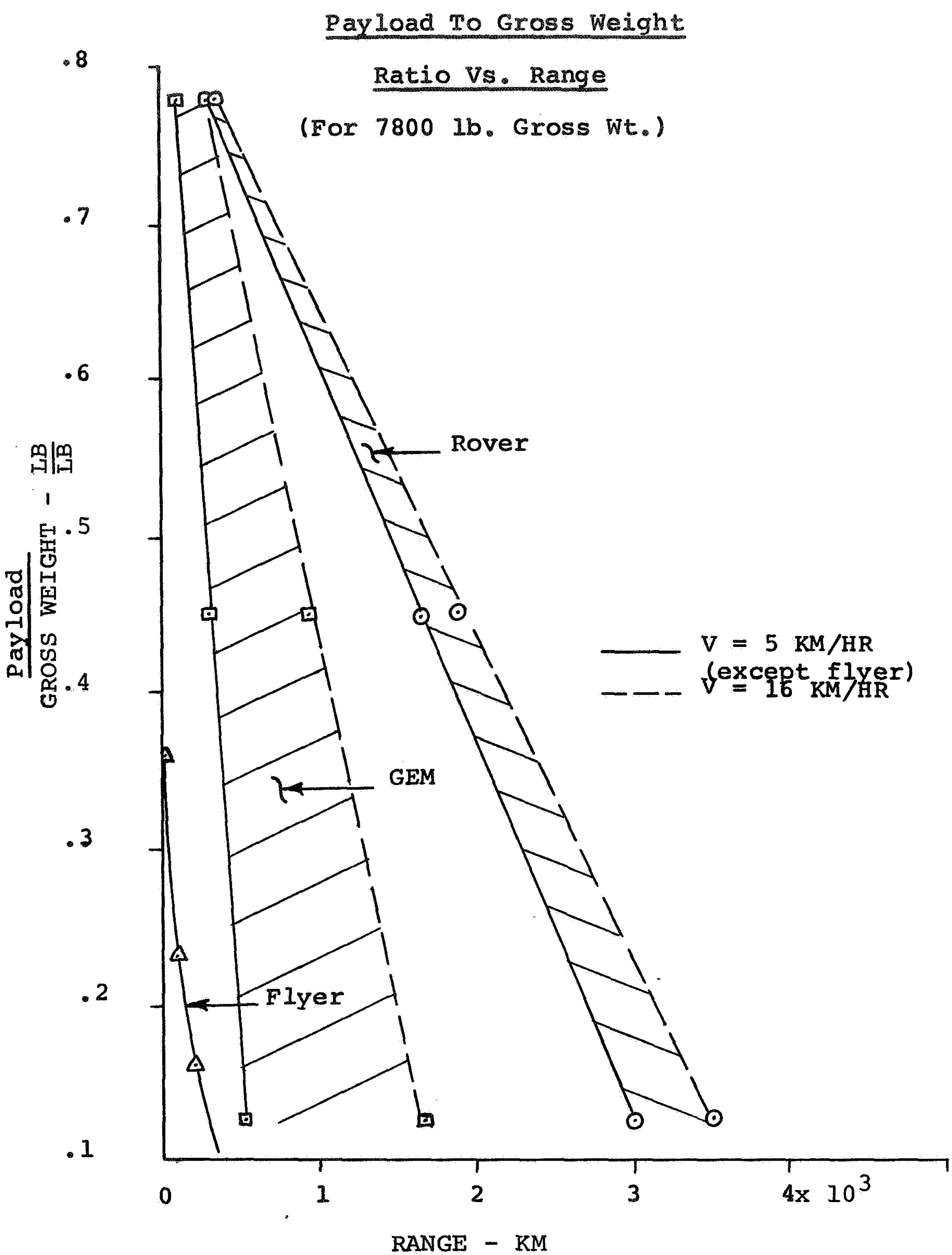

Payload To Gross Weight
Ratio Vs. Range
(For 7800 lb. Gross Wt.)
Rover
GEM
Flyer
V = 5 KM/HR
(except flyer)
V = 16 KM/HR
Payload / GROSS WEIGHT - LB/LB
.8
.7
.6
.5
.4
.3
.2
.1
0
1
2
3
4x 10^3
RANGE - KM

FIGURE 8.1-2

GEM

Fuel Comsumption Vs. Range

(Gross Weight = 1500 lb.)

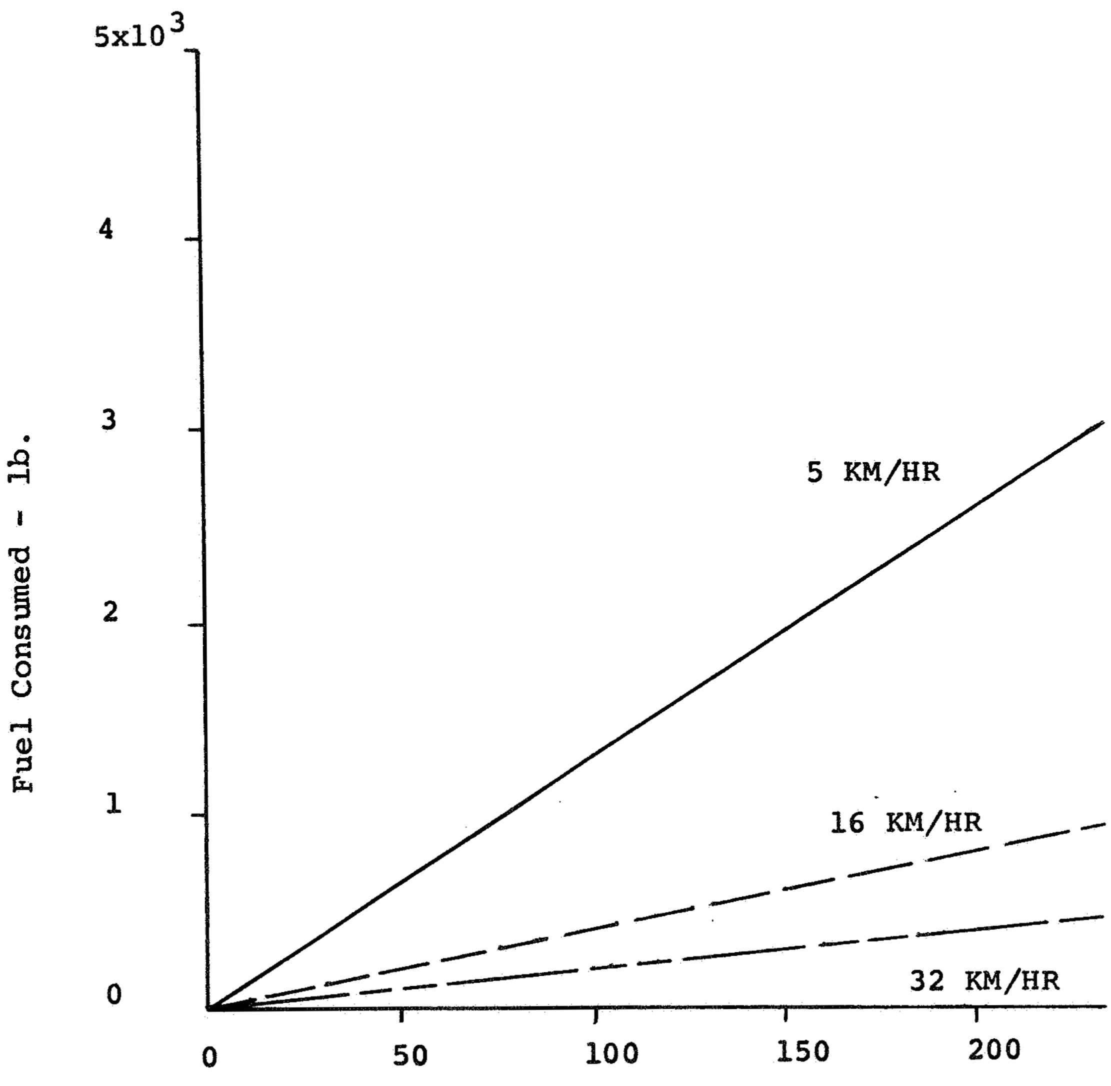


FIGURE 8.1-3

GEM

Fuel Consumption Vs. Range

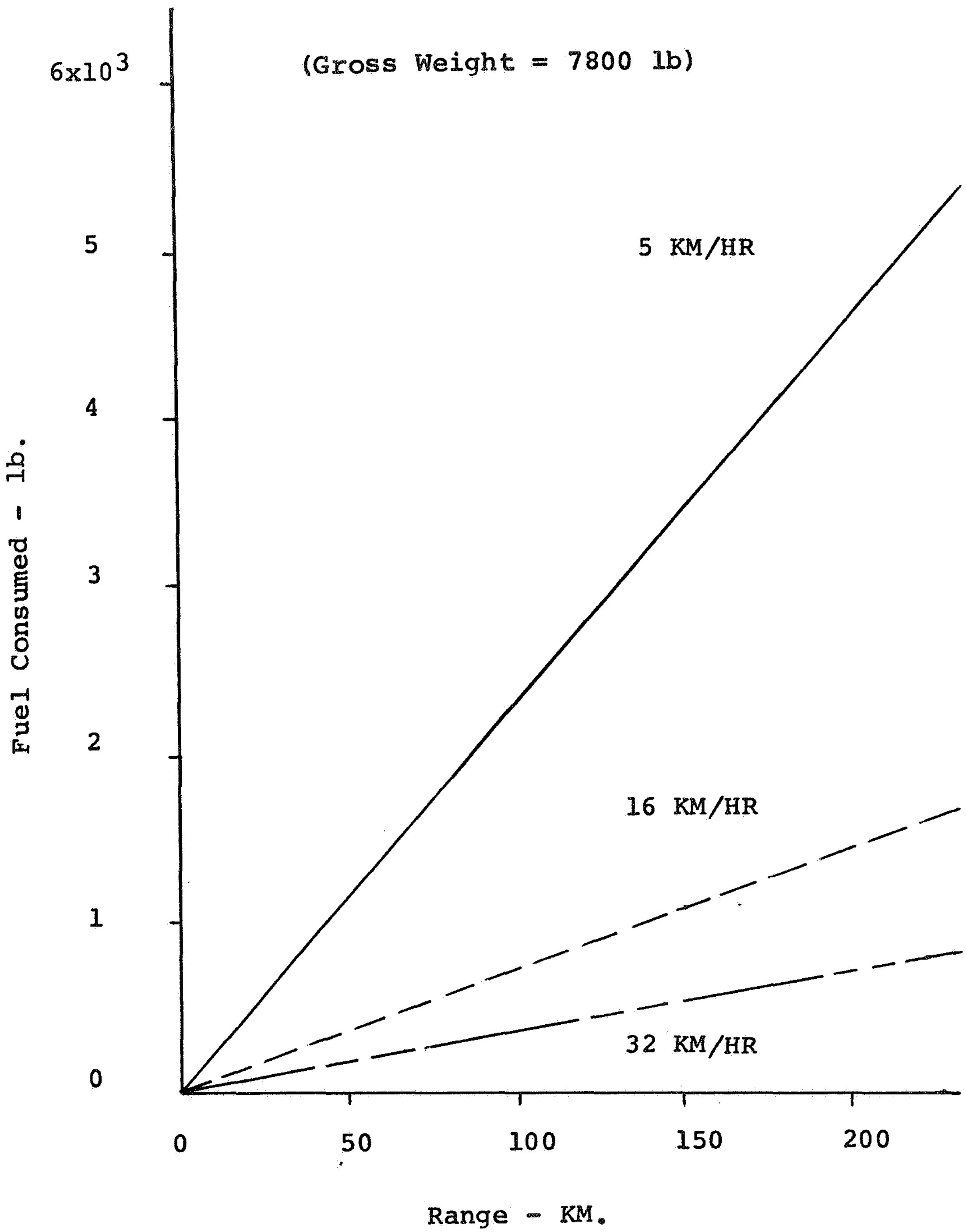
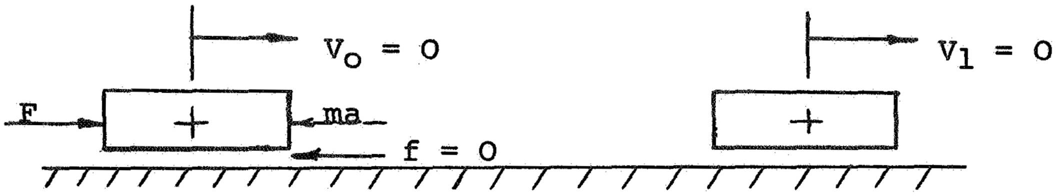


FIGURE 8.1-4



For a rocket propelled vehicle,

$$V = gI_{sp} \ln \left(\frac{W_g}{W_g - W_{fc}} \right) \qquad (8.1\text{-}3)$$

where W_{fc} = Weight of fuel consumed, lb.

V = Attained vehicle velocity, ft/sec

Thus,

$$e^{v/gI_{sp}} = e^k = \frac{W_g}{W_g - W_{fc}}$$

where $k = \dfrac{v}{gI_{sp}}$ Then,

$$W_{fc} = W_g \frac{(e^k - 1)}{e^k}$$

but, the value of e^k in this case is near unity. Therefore, the expansion of e^k will be used.

$$e^k = 1 + k + \frac{k^2}{2!} + \frac{k^3}{3!} + \ldots\ldots\ldots$$

For low values of k, the higher order terms can be neglected. Hence,

$$e^k - 1 = k \text{ and}$$

$$\frac{e^k - 1}{e^k} = k$$ Thus,

$$W_{fc} = k\, W_g \qquad (8.1\text{-}4)$$

With this equation, the fuel consumption was easily calculated for various gross weights, velocities and specific impulses. Strictly for a basis of comparison with other vehicles, an assumption was made of one stop every 5 kilometers. The results are plotted in Figure 8.1-5.

In summary, the analysis of the GEM was based upon several broad and inexact assumptions, in order to provide "ballpark" data during the Phase I parametric studies. Had this class of vehicle been ultimately selected as a prime design candidate, a much more extensive analysis would have been required. However, such was not the case, and the preceding calculations proved to be adequate for comparative purposes.

REFERENCES

1. TRW Proposal No. 13767.000B (1970)

GEM

Ratio of Fuel Consumption With and w/o Stops Vs. Gross Weight

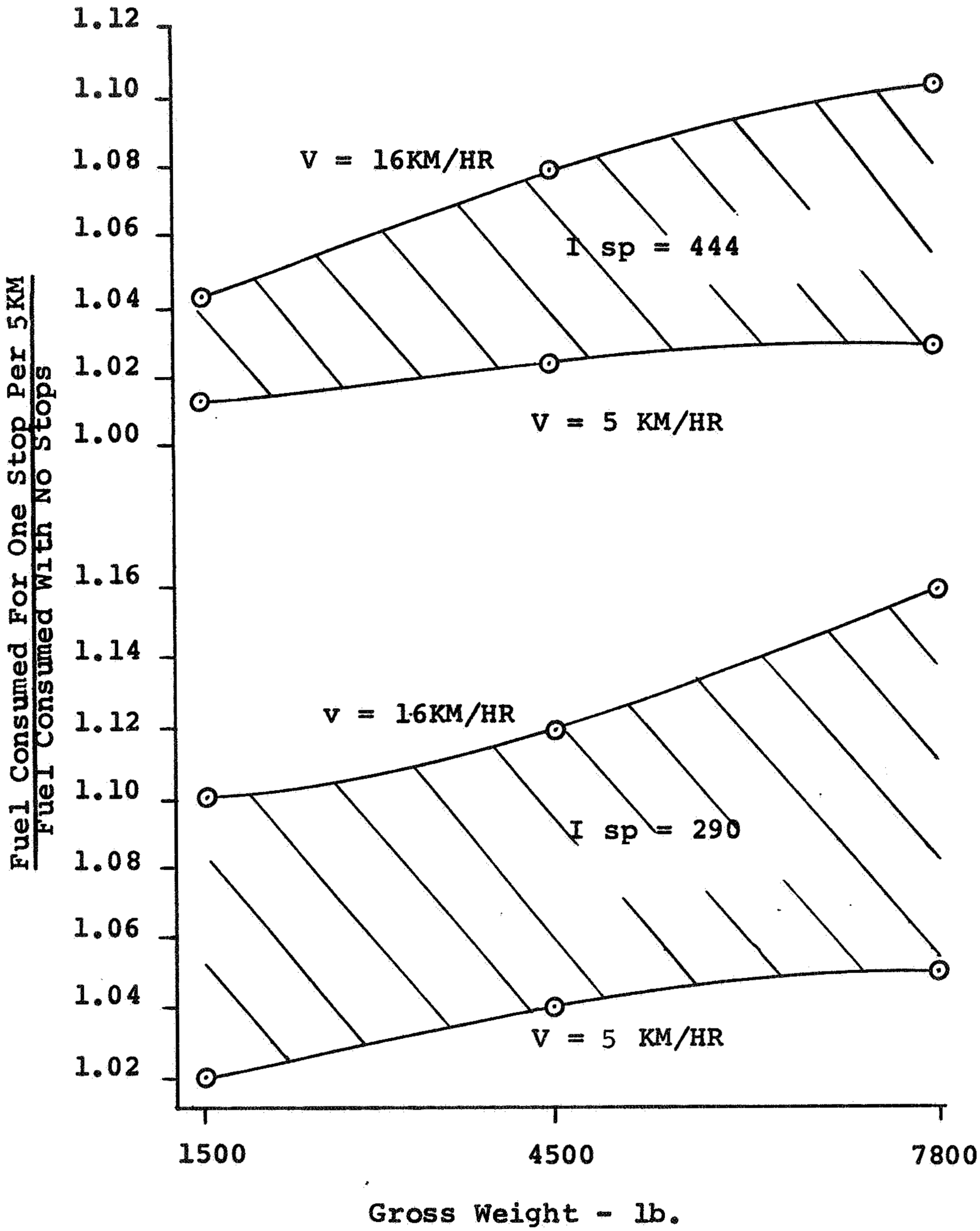


FIGURE 8.1-5

8.1.2. The Flyer

J. M. Ulrich

Another of the mobility system candidates considered was the flyer unit. This type of mobility system is based on an action-reaction principle. Parametric studies were undertaken in order to determine the feasibility of employing a flyer concept within the framework of the gross system requirements.

The equations for the flyer have been determined for a ballistic trajectory. The ballistic trajectory was chosen as a first approximation because it should give the most conservative values for quantities of fuel required for various traversals. If the weight of the vehicle is within reasonable limits, a modified trajectory could then be evaluated.

Using Newton's Law results in the equation

$$m_o \dot{v} = T - m_o g \sin \theta$$

T = thrust, which is assumed constant

$$\dot{v} = \frac{T}{m_o} - g \sin \theta \qquad (8.1\text{-}5)$$

A second equation is obtained assuming the mass of the dry vehicle is equal to the mass at the origin minus the mass burn rate of the fuel times the length of burn.

$$m = m_o - \dot{m}\, t_b \qquad (8.1\text{-}6)$$

where

m = dry mass

m_o = dry mass + fuel mass

t_b = time of burn

Other equations necessary for the solution are

$$v = v_o + a\ t \tag{8.1-7}$$

v_o = initial velocity

a = acceleration

and

$$s = v_o t + \tfrac{1}{2}\ at^2 \tag{8.1-8}$$

Equations (8.1-7) and (8.1-8) can be solved simultaneously to obtain the initial velocity for a given distance and trajectory angle.

Substituting for m_o in Equation (8.1-5) from Equation (8.1-6), an integrable equation is obtained.

After integrating the equation and performing some manipulations, the following equation is obtained:

$$me^{\frac{v_o}{gI_{sp}}} \cdot e^{\frac{(g_m) t_b \sin \theta}{(g_e) Isp}} = m + \dot{m}t_b \tag{8.1-9}$$

This equation can then be solved by a trial and error procedure or by plotting the equation.

The latter method was used in this study for various initial velocities, various values of m, a trajectory with 0-45°, and values for I_{sp} = 290 and I_{sp} = 444. The two values chosen for the I_{sp}'s were selected as two typical ones out of many possibilities to cover a wide range for the specific impulse.

Figures 8.1-6 to 8.1-8 represent typical parametric data developed in Phase I. Results indicate the unmanned mission is completely unfeasible and the manned mission has limited value. Based on the data analysis, the flyer concept was eliminated from considera-

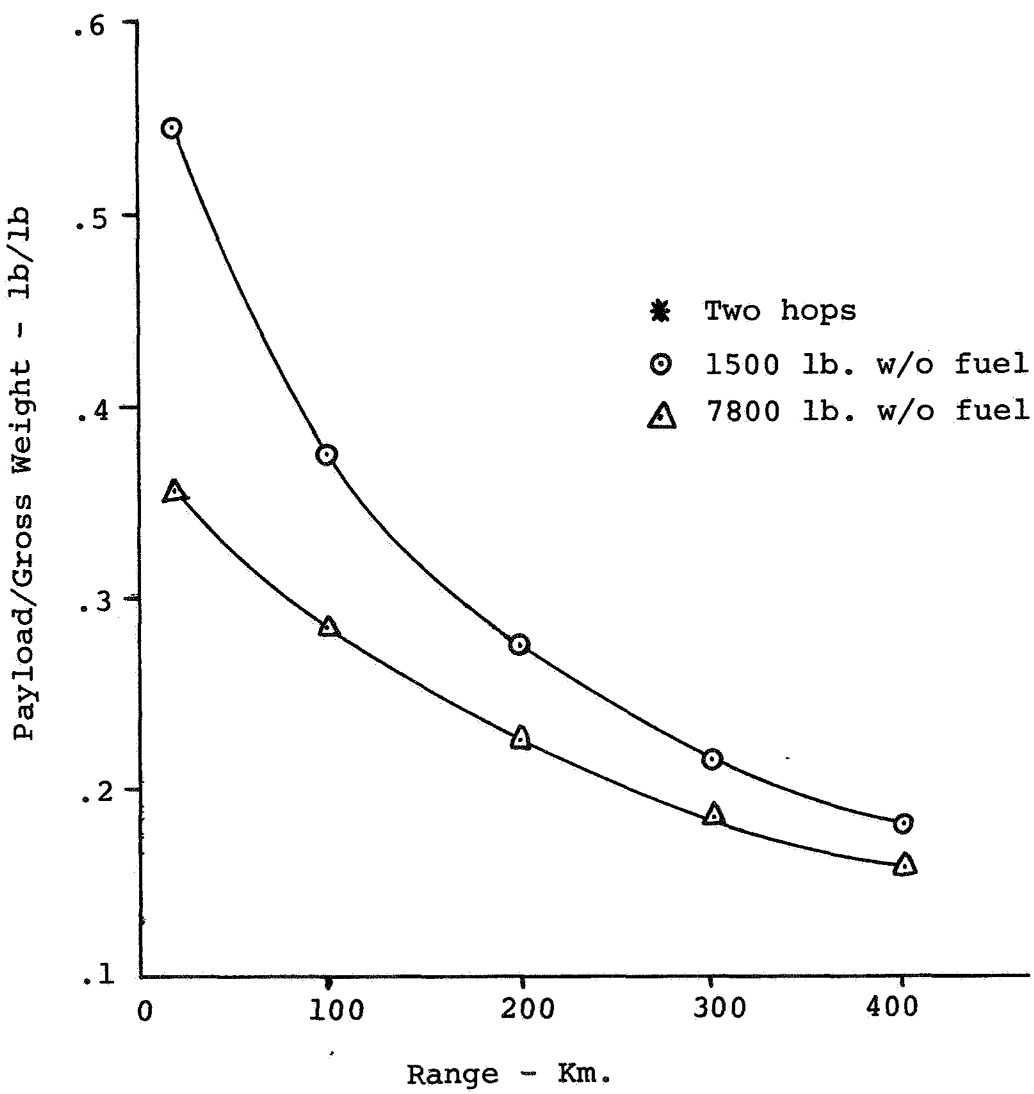


FIGURE 8.1-6 PAYLOAD/GROSS WEIGHT VS RANGE FOR FLYER

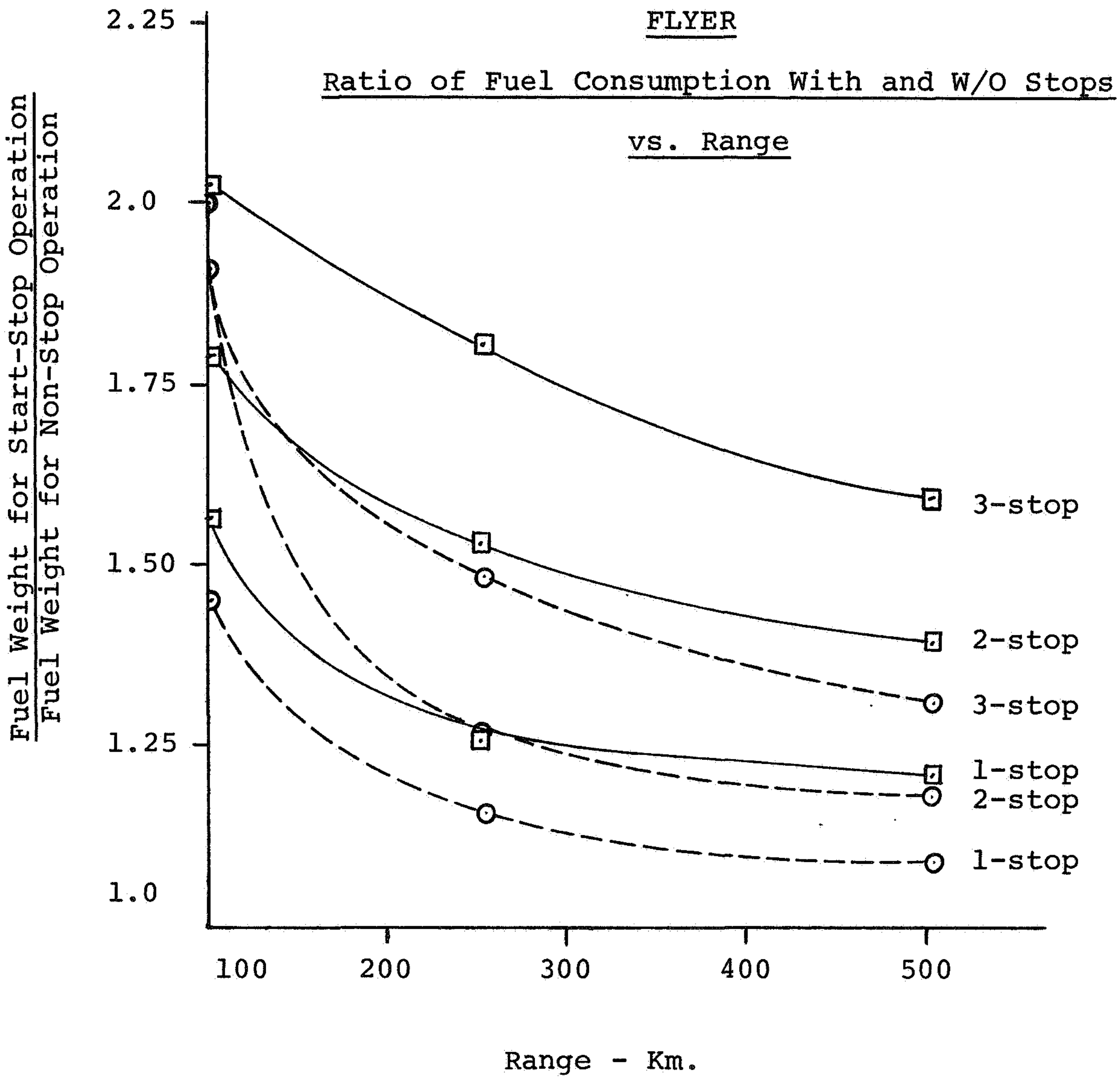


FIGURE 8.1-7 EFFECT OF NUMBER OF STOPS ON RANGE OF FLYER

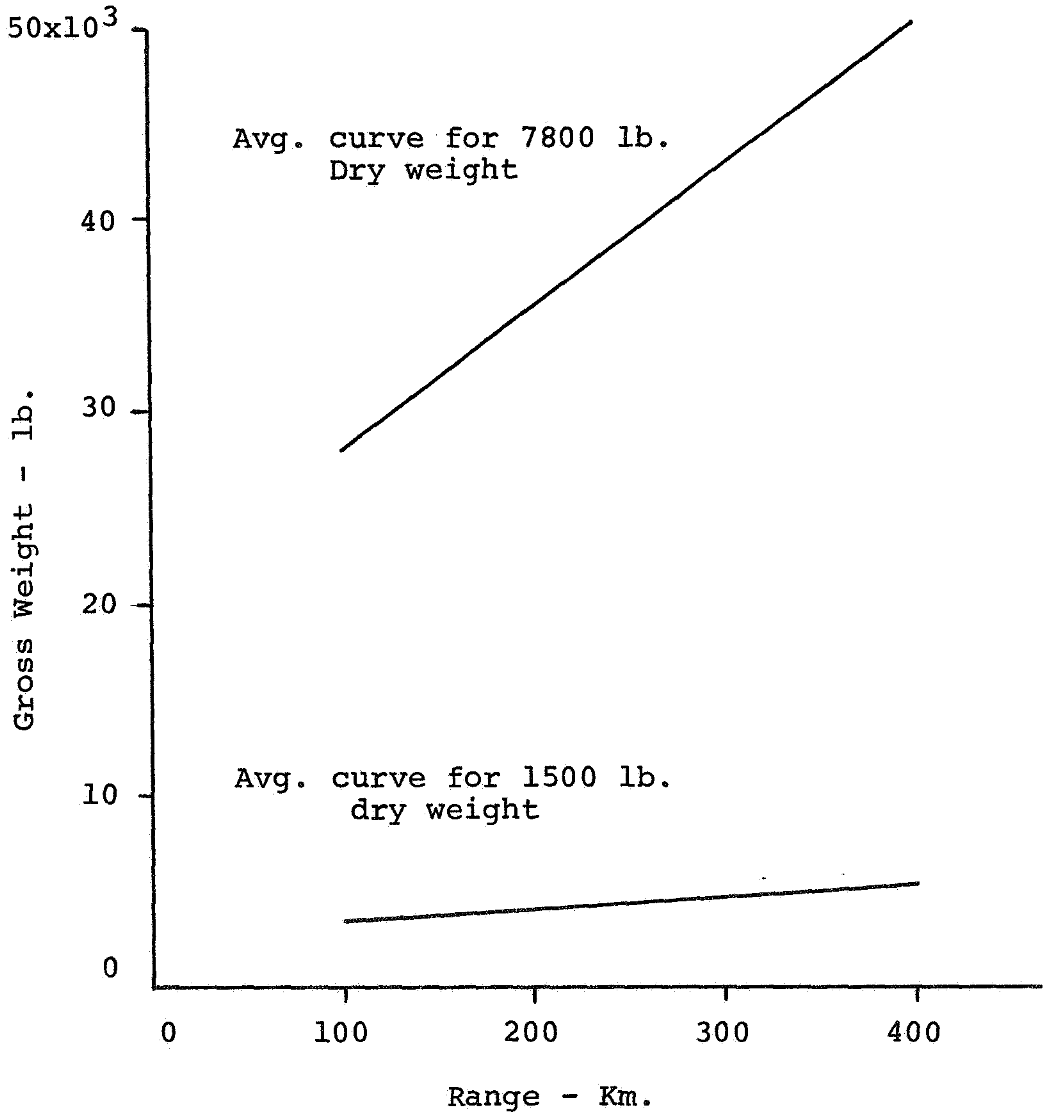


FIGURE 8.1-8 GROSS WEIGHT VS. RANGE FOR FLYER

tion.

8.1.3. Ground Vehicle (Rover)

J. M. Ulrich

A third vehicle class evaluated was the ground rover, basically a 4-wheeled jeep. The analysis was based on the following assumptions for a roving vehicle:

1. uses a power subsystem consisting of RTG, batteries, and fuel cells,
2. travels with constant speed,
3. has an overall propulsion (locomotion) efficiency given by

N = 41%	48%	52%	54%	56%	58%
V = 2km/hr	4km/hr	6km/hr	8km/hr	10km/hr	14km/hr

4. has a specific fuel energy of approximately 1 kwh/lb
5. has four wheels with the following dimensions: b (width) = 10" and D(diameter) = 60", and
6. has a gross weight not more than 10,000 lbs. and not less than 1500 lbs. (earth weight).

We can derive the following representative equations:

$$(\text{Fuel consumption, lb/km}) \equiv U \equiv 1.55 \times 10^{-4} \frac{W^{3/2}}{N} \qquad (8.1\text{-}10)$$

where W is the gross weight of the vehicle and N is the overall efficiency of the locomotion system.

$$(\text{Range, km}) \equiv S = \frac{3.7 \times 10^5}{W^{\frac{1}{2}}} \cdot \left(1 - \frac{W_d+W_p}{W}\right) \qquad (8.1\text{-}11)$$

where W_d = Dry Weight (earth lb.)

W_p = Payload Weight (earth lb.)

From these two equations, payload, gross weight and range were plotted and evaluated relative to the GEM and flyer.

8.1.4. Hopper

J. M. Ulrich

The hopper (Holab) is a unique device proposed as a lunar rover vehicle. Essentially it is a combination ground contact-flyer vehicle and can be pictured by visualizing a pogo stick with a cabin on it.

Two versions of the hopper were analyzed, one of which weighed a total of 1190 pounds with a science payload of 22.5 pounds and a larger version weighing 7380 pounds and a scientific payload of 320 pounds. The hopper was eliminated, based on control and human factors considerations. As an example of the above-mentioned factors, it would require 1380 hops to travel 20 miles, at an average speed of 10 feet/second, with a time delay of 2 seconds between hops and 5.65 seconds for a hop using the 1190 pound vehicle.

8.2. ASTRIONICS EVALUMATRIX

L. C. Ludeman

The parameters considered important by the Astrionics group were the cost, weight, complexity, and reliability of the various subsystems. The subsystems considered were navigation, guidance and control, communication and data processing. The Rover was taken as a base with ten points for each subsystem. The other configurations were ranked comparatively with a lower score meaning inferiority in the subsystem with respect to the defined parameters. From the Astrionics viewpoint there was essentially no difference between the Rover and the GEM as seen in Table 8.2-1. The piggy backs and combinations suffered slightly as duplicate systems were necessary.

TABLE 8.2-1 EVALUMATRIX FOR ASTRIONICS

	Rover	Flyer	GEM	Hopper	Flyer-GEM Hybrid	Rover-Flyer Combination	Rover-GEM Combination	Rover-Rover Piggy Back	Rover-Flyer Piggy Back	Rover-GEM Piggy Back	GEM-GEM Piggy Back	GEM-Rover Piggy Back	GEM-Flyer Piggy Back
Navigation	10	7	10	8	7	7	9	9	5	6	6	6	5
Guidance & Control	10	4	9	1	3	4	8	8	3	8	8	8	3
Communication	10	10	10	6	10	7	7	7	7	7	7	7	7
Data Processing	10	10	10	8	9	8	8	8	8	8	8	8	8
TOTALS	40	31	39	23	29	26	32	32	23	29	29	29	23

8.3. Human Factors

J. T. Emanuel

8.3.1. Explanation of Rating Methodology

The Human Factors Evalumatrix was an attempt to evaluate the prime mobility candidates from the operators' point of view. No attempt was made to evaluate all possible factors; only those directly impinging upon the operator were examined. A complete list of the factors considered is contained in Table 8.3-1.

The various candidates were rated from 0 to 10 for each of the listed factors. In all cases a score of 10 was assigned only when a candidate was judged to be optimal with regard to that factor. In cases when none of the candidates was judged to be "optimal", the top award was less than 10.

Several general criteria permeated the assignment of specific values to the various parameters.

1. In general, for factors that relate to vehicle or astronaut safety, slow vehicles received more points than fast vehicles and ground or near ground vehicles received more points than those that operate above the ground. This philosophy was deemed valid since a prime consideration is preserving the health and well-being of the astronauts. This accounts for the relatively high ratings received by the rover and the relatively low ratings received by the hopper and flyer for safety, reliability, stability, crew training time, crew training cost, probability of

FACTOR	Candidates: Rover	GEM	Flyer	Hopper
Safety	9	7	6	4
Reliability	9	8	8	4
Stability	9	8	5	3
Crew Comfort	6	9	8	0
Crew Involvement in Operation	6	7	6	0
Ease of Control	9	5	6	0
Crew Training Time	10	9	5	3
Crew Training Cost	10	6	4	0
Probability of an Accident	4	2	2	0
Consequence of Accident	9	8	2	0
Unmanned Controllability	4	5	5	0
Time to Ingress/Egress	5	5	5	5
Ease of Ingress/Egress	5	5	5	5
Thermal Controllability	9	9	9	9
Surface Visibility	10	8	2	6
Ability to Accommodate Manipulators	10	4	4	1
Operator Decision Time	10	7	4	0
Repair Simplicity	9	7	6	2
Ability to Accommodate Cabin	10	8	8	10
Operator Dependence on Auto. Guidance/Control Systems	9	7	5	5
Mission Time/Operation Time	0	7	10	8
Operator Involvement in Maintenance	5	5	5	5
TOTAL	167	138	120	70

TABLE 8.3-1 HUMAN FACTORS CRITERIA FOR EVALUATION

an accident, surface visibility, ability to accommodate manipulators, operator decision time, and the operator dependence on automatic guidance and control systems (ability to travel without such systems).

2. Parameters such as ease of ingress/egress that were considered to be design dependent, rather than vehicle dependent, were assigned a constant value for all vehicle systems. This action was taken since we felt that in the final vehicle design these factors should be considered in greater depth because of their potential influence on human performance.

8.3.2. Discussion of Results

The Rover[1] was judged to be the best candidate for the following reasons.

1. The top speed of the rover is approximately 15 km/hr. At this speed any accidents should not be totally incapacitating for either the vehicle or the operator.

2. Since the Rover is a ground vehicle any operational failures in the power or propulsion systems should result in a nonoperational vehicle, but should not cause bodily injury to the astronauts.

3. The Rover should require relatively short training periods since the basic operational controls do not differ greatly from those required in operating earth bound vehicles such as automobiles or small tractors.

4. Due to the rover's slow speed, the operator has adequate time to visually inspect the surface characteristics and make control

decisions altering the traverse.

The Ground Effects Machine (GEM)[2] was judged to be a less desirable candidate than the Rover, but better than the other two prime candidates.

1. The air cushion suspension system of the GEM results in a high level of crew ride comfort.
2. The GEM operates close to the ground which gives the vehicle the favorable characteristics discussed under the Rover (statement No. 2).
3. The top speed of the GEM is considerably above 15 km/hr, which would result in less driving time, but would also result in potentially more severe accidents. Most of these accidents should be survivable, but, nevertheless, serious.
4. Due to the inherent dust problems associated with a GEM, the astronaut will have decreased surface visibility.

The Flyer[3] was downgraded because of the following factors.

1. The training time and cost associated with learning to operate the vehicle would be excessive, possibly approaching the values associated with learning to operate the LUNAR MODULE.
2. The top speed of the Flyer is such that the surface visibility would be very poor, the probability of an accident being survived is reduced, and the operator only has a minimal time to make control decisions.

The Hopper[4] was judged to be the least desirable candidate based on some extreme conditions to which the astronauts would be subjected.

1. The maximum accelerations to which the operator would be subjected (about 5 g's) are very acceptable for short durations without repetition. The Hopper imposes values of such a magnitude on the operator as often as once every 8 to 10 seconds. This is totally unacceptable from a human factors viewpoint.[5]

2. Controlling the vehicle requires that the operator visually determine characteristics of the lunar surface at distances of from 10m to 100m. Given the visual problems associated with the lunar surface (See 13.4), these requirements exceed the capability of the unaided eye.

3. The sequence of motor activities required of the operator includes making precise control adjustments before every hop: approximately every 8 seconds. This level of activity is physically and mentally exhausting for the operator.

In summary, from a human factors evaluation, slow surface vehicles are good, and fast off-the-surface vehicles are judged to be poor. Those vehicles having characteristics between these extremes are judged to be of intermediate value as lunar mobility candidates.

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1. "Apollo Logistics Support System (ALSS) Payloads", MOLAB NASA, G. C. Marshall Space Flight Center. Contract NAS8-11411, June 1965.

2. Unsolicited Proposal - Lunar Ground Effects Machine. TRW Systems, October 1969.

3. "One-Man Lunar Flying Vehicle Final Report", Phase B Study 7335-950012 Bell Aerospace Company, Contract NAS9-9044 (MSC) July 1969.

4. Kaplan, M. H., Dynamical Equations for the Plane Change Maneuver of the Lunar Hopping Laboratory. J. Spacecraft and Rockets Vol. 7, No. 4, April 1970.

5. Handbook of Human Engineering Design Data for Reduced Gravity Conditions. Marshall Space Flight Center Contract NAS8-18117, Manned Spacecraft Center Contract NAS9-8640. General Electric, Space Systems Organization.

8.4. Configuration Evalumatrix

J. C. Lindholm

The configuration group was concerned with the overall mobility system. Thus in developing a set of parameters for the evalu-matrix a much larger set resulted than in the other matrices. Some of the parameters selected for the evaluation were in the specific areas of the other three groups. The matrix used in the evaluation consisted of twenty-two parameters and was used to evaluate all thirteen mobility system candidates.

Before working on the evaluation criteria, studies of reports on other mobility concepts that had been prepared for or by NASA - MSC were made. After these studies, I decided to generate a list of parameters to use. A tentative list was made following discussions with members of the team. Two categories of parameters were developed. The first was a list of possible mission tasks for exploration of an extraterrestrial body. After all, before a design can be selected, the functions that it is to perform must be spelled out in detail. These tasks are indicated in Table 8.4-1 and are not the same as given in Chapter 5. It was assumed that most of the missions would be directed toward study of the moon surface layers. Later missions might be directed toward observations of the earth and other space bodies from a moon base. These tasks were weighted as to importance and this is shown in the table also.

TABLE 8.4-1. PROPOSED MISSION TASK EVALUATION CRITERIS

Mission Task Parameter	Weight Factor
Surface Sample Collection	8.0
Selenodetic Surveys	7.5
Subsurface Sample Collection	7.4
Astronomical Observations	6.7
Seismic Measurements	6.3
Supply Hauling	5.5
Establish Lunar Base	5.0

The second category of parameter are those parameters that affect the safety and comfort of the crew or the performance of the mobility system. These are given in Table 8.4-2.

The mobility system parameters were weighted and the values are indicated in Table 8.4-2. These lists were presented to members of the team with a specific request for additional parameters. The members of the configuration group were asked to weight each parameter from 0 to 10 as to its importance in the overall accomplishment of exploration. A weight of 10 indicates a very important parameter.

A careful study of the parameters showed that some were interdependent and thus provided essentially the same information. To accomplish the mission tasks it was decided that the mobility system had to be able to do some particular function. For

TABLE 8.4-2 PROPOSED MOBILITY SYSTEM EVALUATION CRITERIA

Mobility System Parameter	Weight Factor	Mobility System Parameter	Weight Factor
Safety	9.3	Reliability	9.0
Stability	8.3	Payload Weight	8.2
Terrain Negotiability	8.0	Range	7.9
Start/Stop Ability	7.6	Fuel Weight	7.6
Operational Simplicity	7.4	Reuse	7.0
Maintenance	6.8	Crew Comfort	6.8
Cabin Weight	6.8	Life Support Equip. Wt.	6.7
Total Weight	6.7	Power Pack Wt/Kw Output	6.5
Speed	6.4	Night Operation	6.2
Dry Weight	6.2	Commonality	5.9
Number of Men Carried	5.8	Manned/Unmanned Conversion	5.5
Number of Scientific Stops per Mission	5.5	Crew Training	5.4
Lifetime	5.4	Remote Manipulators	5.0
Overall Size	5.0	Bulldozer Capability	3.0

instance surface sample collection can be done by an astronaut during an EVA period or by remote manipulators from within the cabin. Thus keeping both of these factors would doubly favor the system that could do this best. The final list of parameters and their weight factors are given in columns 1 and 2 of Table 8.4-3.

The evaluating of the proposed mobility system candidates was done by each member of the configuration group. The values placed on a particular parameter for a certain candidate were averaged to get the value assigned to that place in the evalu-matrix. After all the ratings had been averaged the values for each candidate were summed to give the total points shown at the bottom of Table 8.4-3. These values were normalized by assigning the rover, which had the most total points, a value of 100. Each of the other candidates was given a percentage of 100 based on the total points it received. These values are also given in Table 8-1 for comparison with the results of the other groups.

TABLE 8.4-3 CONFIGURATION EVALUMATRIX

Performance Parameters		Primary Candidates				Secondary Candidates: Hybrid Combinations			Secondary Candidates: Piggy Back					
	Weight Factor	Rover	Gem	Flyer	Hopper	Flyer-Gem	Rover & Flyer	Rover & Gem	Rover on Rover	Flyer on Rover	Gem on Rover	Gem on Gem	Rover on Gem	Flyer on Gem
Safety	9.3	87	63	22	24	35	49	77	88	51	77	70	77	47
Reliability	9.0	83	70	56	41	63	68	74	81	79	81	68	77	65
Unmanned Controllability	9.0	75	39	41	5	22	58	59	72	63	59	32	34	29
Stability	8.3	81	62	32	10	37	58	64	77	58	68	54	58	50
Payload Weight	8.2	77	51	21	28	25	74	78	70	68	70	43	43	37
Terrain Negotiability	8.0	46	50	66	48	66	66	60	54	66	64	56	56	66
Start-Stop Capability	7.6	71	48	25	34	44	62	67	71	68	70	38	41	38
Operational Simplicity	7.4	69	46	25	21	19	30	46	64	31	44	37	43	19

TABLE 8.4-3 CONFIGURATION EVALUMATRIX (CONTINUED)

Performance Parameters		Primary Candidates				Secondary Candidates: Hybrid Combinations			Secondary Candidates: Piggy Back					
	Weight Factor	Rover	Gem	Flyer	Hopper	Flyer-Gem	Rover & Flyer	Rover & Gem	Rover on Rover	Flyer on Rover	Gem on Rover	Gem on Gem	Rover on Gem	Flyer on Gem
Range	7.0	60	52	44	35	57	77	69	59	75	67	47	43	57
Reuse	7.0	67	63	60	54	56	60	60	64	59	60	55	59	55
Versatility	7.0	47	29	28	16	38	66	49	46	67	50	28	35	48
Maintenance	6.8	60	44	40	33	33	45	48	57	46	48	33	33	29
Crew Comfort	6.8	57	53	37	12	41	42	48	52	52	50	50	48	42
Total Weight	6.7	45	56	47	54	31	25	42	42	35	39	46	40	39
Speed	6.4	20	47	61	36	58	47	36	22	41	35	44	42	49
Night Operation	6.2	51	30	20	11	20	45	46	51	45	45	26	32	24
Commonality	5.9	51	27	34	13	21	44	34	50	41	33	24	33	24

TABLE 8.4-3 CONFIGURATION EVALUMATRIX (CONTINUED)

Performance Parameters		Primary Candidates				Secondary Candidates								
						Hybrid	Combination		Piggy Back					
	Weight Factor	Rover	Gem	Flyer	Hopper	Flyer-Gem	Rover & Flyer	Rover & Gem	Rover on Rover	Flyer on Rover	Gem on Rover	Gem on Gem	Rover on Gem	Flyer on Gem
Manned/ Unmanned (Conversion)	5.5	47	19	23	6	14	37	36	41	36	35	20	21	15
Crew Training	5.4	50	30	18	10	14	21	28	49	24	31	24	27	14
Lifetime	5.4	49	38	32	18	36	41	41	45	40	41	31	33	30
Remote Manipulators	5.0	46	31	23	28	30	43	44	46	44	45	27	32	25
Bulldozer Capability	3.0	29	8	3	1	4	28	28	29	28	28	6	13	3
Total Points		1268	960	754	538	764	1086	1134	1230	1117	1140	859	920	805
Normalized Value		100	76	60	43	60	86	90	98	89	90	68	73	63

CHAPTER 9

EFFECTIVENESS COST STUDY

J. E. Sneckenberger

The present chapter is concerned primarily with presenting a methodology that was developed for evaluating candidate mobility systems. In contrast to the comparative evaluation technique used in the previous chapter, this evaluation method examines each candidate on an individual basis. The various mobility system candidates examined can then be ranked in order of desirability according to a proposed criterion. The order of ranking of the alternatives can thus be considered when asked to select the most promising candidate.

While the method developed resembles in some respects the cost effectiveness approach often used by system analysts, it should be pointed out that the methodology developed was conceived principally for application in performing the task of selecting a class of mobility systems. No serious attempt was made to apply the principles of cost effectiveness in a rigorous manner.

9.1. Objectives of Evaluation Procedure

Before beginning a discussion of the evaluation procedure developed, a few comments should be made related to the general philosophy which led to this method of evaluation. One of the goals of the

Evaluation Committee was to develop an evaluation procedure which would examine a particular candidate mobility system in terms of its ability to satisfy the gross system requirements of Chapter 5. In as much as this conceptual study of mobility systems was mission oriented, the committee originally endeavored to develop a procedure which would evaluate each alternative mobility system solely for its ability to perform the missions specified in the gross system requirements. However, the need for including a second measure of a system's design into the decision-making process was soon recognized. (One obvious instance of the need for a second measure of a system's design is to distinguish between two systems of equal ability.) Consideration was given to such system design measures as reliability, survivability, and maintainability. These measures were considered for the purpose of avoiding the uncertainty usually associated with cost estimates of future generation space vehicles. However, in all such considerations, the value in quantifiable terms for each suggested measure of system design was either closely related or easily transformed to a cost estimate. It was also concluded that realistically some form of cost estimate must be considered as the second measure of a system's design.

The two important objectives which thus evolved and which directed the development of this evaluation procedure are best expressed by the following two questions: How well does a candidate mobility system perform the required missions?, and, how much does it cost? The quantitative calculation of "how well" and "how much" will be presented in Sections 9.3 and 9.4, respectively.

Having realized a need for a second measure of a system's design in the evaluation of alternatives, a reasonable extension might be to consider the contribution a third (or even fourth) measure of a system's design would introduce to the usefulness of the evaluation methodology. This appraisal, however, was considered beyond the scope of this study.

In these conceptual phases of a systems study such as this, there may be a very large number of alternative systems to be considered. The problem is to screen these alternative systems by a selection procedure, with the intent of weeding out the obvious unattractive systems. However, neither the need nor the time for the precise definition of such a procedure existed during this study.

9.2 Formulation of Methodology

In the development of a methodology for evaluation of mobility system concepts for future planetary exploration, an important characteristic which should be inherently incorporated into the methodology is the potential for subsequent reutilization of the methodology. (Of course, updating of input data, such as extended mission profiles, new concepts in mobility systems, etc. is to be expected, but the methodology hopefully should remain unchanged.) Therefore, an attempt was made to develop a general formulization of the evaluation procedure which might be applicable to future studies. (See Figure 9.2-1).

At the time of this study, the scope of the development of planetary mobility systems concepts was roughly as follows: A prototype

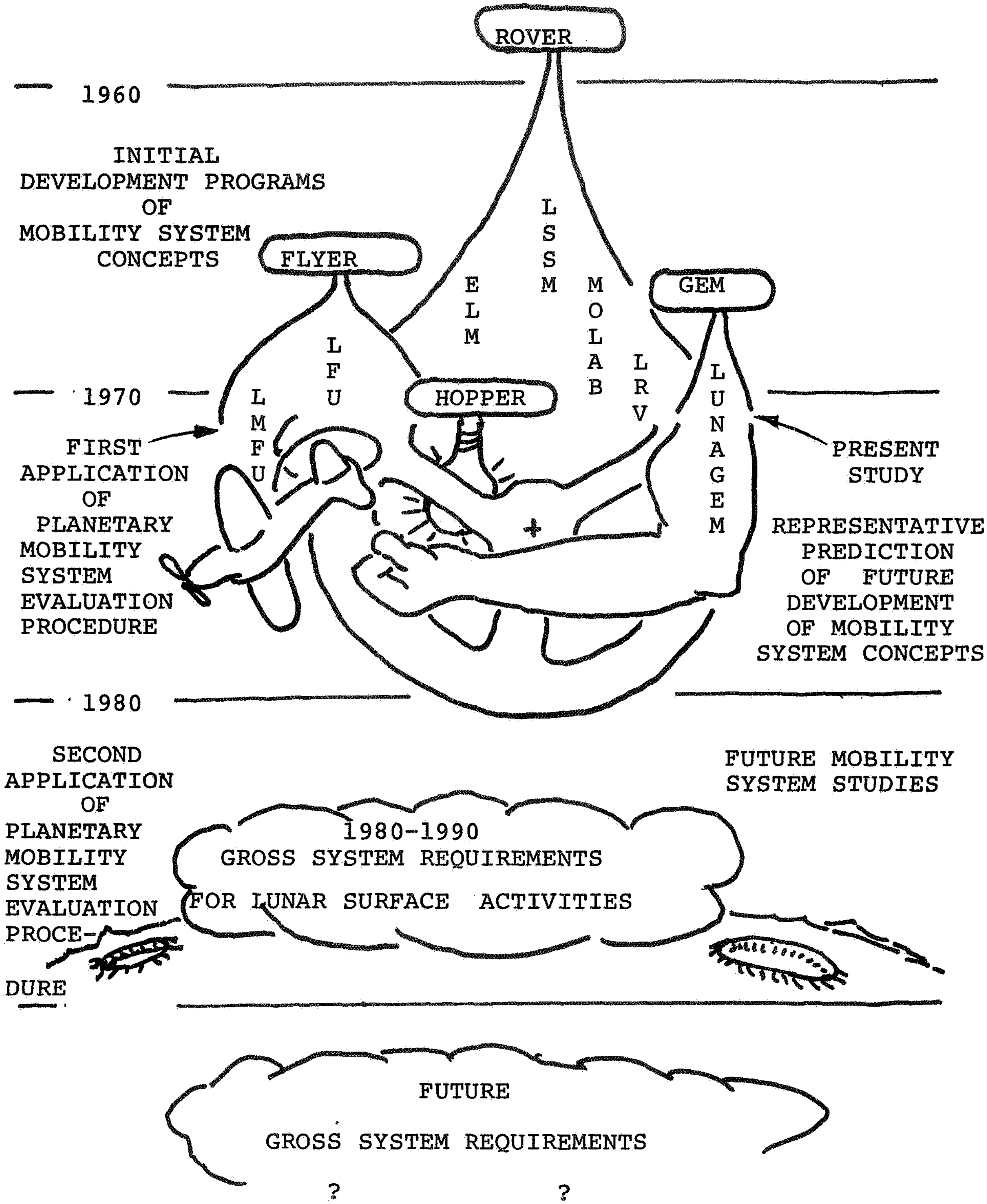

FIGURE 9.2-1 FUTURE APPLICATION OF METHODOLOGY

lunar roving vehicle (LRV), designed to provide lunar surface mobility for a future Apollo landing, was presently being evaluated. The one-man versions of the lunar flying unit (LFU) and the lunar ground effects machine (LUNAGEM), while still basically in the conceptual design stage, were still being considered for lunar mobility application. Several other novel mobility concepts had also been proposed. Hence, the available mobility system descriptions, subsystem definitions, etc., while quite meager, did provide the baseline sketches of system concepts upon which to formulate an evaluation procedure for the general mobility system. As the space program, and systems for operation in the lunar environment in particular, gains further experience, planetary mobility system definition and design will become more descriptive. This will permit a higher level of evaluation among the alternative concepts. In terms of the present, it was the goal of the Evaluation Committee to formulate the evaluation methodology and to use the procedures developed to aid in the conceptual design of a mobility system which would most effectively perform the specified missions at the least cost.

In the following two sections of this chapter, the general description of the effectiveness model and the cost model are presented. Once formulated, these models provide the means by which two measures of a candidate mobility system can be computed that are useful in ranking it with other alternatives. The scheme for displaying these measures of a candidate system is shown in Figure 9.2-2. Each candidate is identified with a point on the graph, while variations within a class of mobility system can be

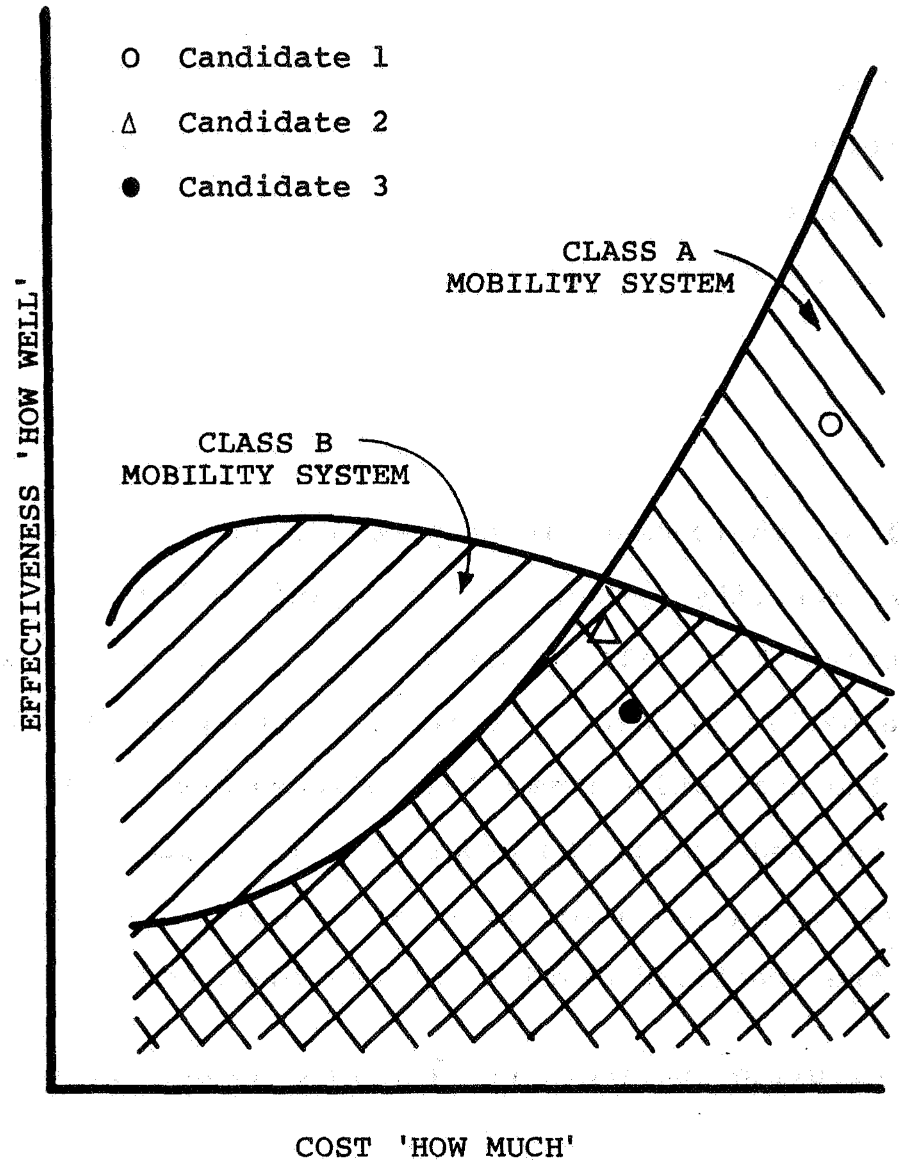

FIGURE 9.2-2 GENERAL EFFECTIVENESS COST GRAPH

represented as areas of the graph. The criterion for ranking the various alternatives can take several forms. If maximum effectiveness were the criteria, candidate 1 is obviously the desired alternative. For a limited budget of less than that of candidate 1, candidate 2 is preferred over candidate 3. Very likely there will be an established minimum acceptable effectiveness as well as an upper constraint on the budget available.

9.3 Measure of Effectiveness

The concept of effectiveness, as applied to mobility systems, deals primarily with a system's ability to perform surface activities in the lunar environment within a stipulated period of time. In Chapter 5, the broad definition of lunar surface activities requiring the use of a mobility vehicle was converted into typical mission categories with corresponding task descriptions. In order to develop an effectiveness model for this study which would facilitate the systematic calcuation of a single quantitative measure of effectiveness for each candidate mobility system, various mission parameters, such as payload involved, total distance traversed, etc., were identified. Of these parameters, those that were useful in expressing the capability of a mobility system to perform the specified missions were incorporated into an effectiveness model. Table 9.3-1 identifies these useful parameters along with their numerical values for each mission.

Table 9.3-2 serves as a convenient means by which the tasks involved on each mission are easily identified when evaluating a

TABLE 9.3-1. MISSION PARAMETERS

MISSIONS		MISSION PARAMETERS		
NO.	DESCRIPTION	TIME MOVING (HRS)	TOTAL DISTANCE (KM)	PAYLOAD* (LB)
I A	Rendezvous	336	1500	0
I B	Drill	216	1000	1670
I C	Stations	312	1300	1200
II	Science	312	1400	500
III	Science	26	250	500
IV	Traverse	16	250	1870
IV A	Traverse/Sortie	4	5	80
V	Sortie	4	5	80
VI	Support	12	10	0
VII	Support	168	300	0
VIII	Supply	1	6	2000
IX	Supply	6	6	2000
X	Transport	1	6	2000

* Excluding Crew

TABLE 9.3-2. MISSION PROFILES USEFUL FOR EVALUATION

MISSION 1: UNMANNED LONG TRAVERSE

1A. Rendezvous with Tug (distance)

1B. Moving, drilling 2 holes and collecting samples (distance, payload)

1C. Deploying science stations and moving to rendezvous (distance, payload)

MISSION 2: UNMANNED LONG TRAVERSE

Unmanned automatic collection and transmission of geophysical data, surface sample collection (distance, payload)

MISSION 3: MANNED MEDIUM TRAVERSE

Collection of scientific experimentation data, transmission of data to tug. (time in traverse, distance, payload)

MISSION 4: MANNED MEDIUM TRAVERSE

Deploy drill or science stations, collect rocks, collect core samples (distance, payload)

MISSION 5: MANNED SHORT SORTIES

Brief sorties conducted from tug or a brief excursion from the main longer traverse (moving time, total time in traverse, distance, payload)

MISSION 6: MANNED LUNAR BASE ESTABLISHMENT

Assist in establishing lunar base, grade soil. (Can it be used to move soil?)

MISSION 7: UNMANNED LUNAR BASE ESTABLISHMENT

Same as Mission 6 - only unmanned (Can it be used to move soil?, distance)

MISSION 8: MANNED BASE SUPPLY

Supply lunar base with fuel, equipment, and maintenance. (time, payload, distance)

TABLE 9.3-2. MISSION PROFILES USEFUL FOR EVAULATION (CONTINUED)

MISSION 9: UNMANNED BASE SUPPLY

Same as Mission 8 - only unmanned
(time, payload, distance)

MISSION 10: PERSONNEL TRANSPORT

Manned transport of personnel between lunar base and tug.
(time, payload, distance)

candidate system. The words in parentheses identify the function or parameter(s) which a candidate system should possess in order to be effective for that particular mission.

Algebraically, these useful parameters can be expressed as an adequacy number, A_{ij}, as shown by the following equation:

$$A_{ij} \equiv \text{Adequacy of jth candidate to perform the ith mission}$$

$$= \frac{1}{3}\left[\frac{\text{Available Time Moving}}{\text{Expected Time Moving}} + \frac{\text{Possible Distance Traversed}}{\text{Required Distance Traversed}} + \frac{\text{Vehicle Payload}}{\text{Specified Payload}}\right] \qquad (9.3\text{-}1)$$

Each term in the right hand expression is stipulated to be equal to or less than one. Thus, A_{ij} is a number between 0 and 1, and represents the percent effectiveness of the jth candidate system, S_j, in performing the ith mission M_i.

The measure of effectiveness of a candidate system to perform all the specified missions might be determined by summing the candidate's adequacy to perform each mission. However, certain of the missions contribute more toward achievement of the space program's goals of planetary exploration. Thus, each mission was weighted to reflect its relative importance in achieving these goals, some of which are (1) scientific exploration of the lunar surface, (2) exploitation of lunar material, (3) improved space technological capabilities, and (4) furtherance of planetary exploration. Hence, the effectiveness of S_j, determined by summing the candidate's weighted adequacy to perform each mission, is easily expressed in equation form as:

$$E_j \equiv \text{effectiveness of } S_j \text{ as a mobility system}$$

$$= \sum_{i=1}^{n} W_i A_i \qquad (9.3\text{-}2)$$

where

n - number of specified missions

W_i - weight value of M_i such that

$$0 \leq W_i \leq 1 \quad \text{and} \quad \sum_{i=1} W_i = 1$$

It can now be seen that, having been given the necessary information pertaining to the speed, range, and payload of any candidate mobility system, it is then possible to calculate, using Equation (9.3-1), the adequacy of the candidate in performing each of the specified missions, and then to determine, using Equation (9.3-2), "how well" (i.e., its effectiveness) the candidate can perform the specified lunar surface activities.

It should be noted that the effectiveness model can also serve as a tool by which the sensitivity of the effectiveness of a candidate system to changes in the values of the parameters (range, payload, etc.) can be evaluated. It is also important to note that system effectiveness has purposely been developed void of any direct implication or connection with system cost.

9.4. Measure of Cost

In conceptual studies, especially when only the relative abilities of various alternatives is of prime importance, it is convenient to consider comparative cost rather than total mobility system program cost. A full identification of the timing, long-range implications, etc. of procuring a given quantity of mobility vehicles, involving facilities and support equipment, acquisition and training of crew, and a host of other related items is not generally necessary for decision-making purposes in conceptual studies.

Therefore, the basic cost model developed for this study was that of the comparative cost estimate. Considered primarily as indexes, differences in system cost estimates then indicated basic cost increment between alternative mobility systems. Thus, the model reflected only those costs associated with the major hardware equipment, the time involvement of crew, the necessary consumables to perform the missions, etc., which had a direct source impact between alternative mobility systems.

The cost factors that contributed most toward sensitivity between alternative mobility systems were; (1) the cost of developing the vehicle, (2) the vehicle production cost for manufacture of the necessary number of vehicles, (3) the cost of deploying the vehicles on the lunar surface from a weight penalty standpoint, and (4) the cost of operating the vehicles as part of the 1980-1990 period of lunar surface activities. This last factor item (4), is a function of both the expended consumables (fuel, food, etc.) and the manned mission activity (manhours) which requires crew participation. Inherent in these costs are the following cost elements:

(a) Cost for reasearch, development, testing, and evaluation (RDT&E),

(b) Cost of the first flight-ready unit,

(c) Cost to deploy one pound mass on the lunar surface in the 1980-1990 period, and

(d) Cost for support from mobility crew related to mission activity per hour in the 1980-1990 period.

In making estimates of these costs, the price level prevailing in 1970 will be used, i.e. projected costs for the 1980-1990 period will not be inflated. This will permit maximum reliable use of cost data collected from past studies of similar systems and programs.

The measure of comparative cost for a candidate mobility system can be described in equation form as follows: defining, as previously, for the effectiveness model, the indicies i and j such that

i represent missions, $i = 1, 2, \ldots, M$, and

j represent candidates, $j = 1, 2, \ldots, p$

and let

M_i be a description of the ith mission, and

S_j be a description of the jth candidate.

For costing purposes, also let the index

k represent subsystem cost categories, $k = 1, 2, \ldots, q$.

and let

Q_k be a description of the kth subsystem cost category.

(Note: Cost categories are defined in Chapter 20).

Further, define

$B_j \equiv$ Number of S_j built for test and evaluation,

$D_j \equiv$ Number of S_j deployed during 1980-1990

m_{jk} ≡ Mass of Q_k on S_j, dry weight

m_j ≡ Mass of S_j dry weight, $= \sum_{k=1}^{q} m_{jk}$

N_i ≡ Number of M_i during 1980-1990

c_{ij} ≡ Mass of consumables used on M_i by S_j

h_{ij} ≡ Astronaut time used on M_i by S_j, and

$\$_{jk}$ ≡ Cost of Q_k on S_j per unit mass

In addition, let

P ≡ Cost to deploy one pound mass on the lunar surface in the period 1980-1990

R ≡ Cost of one hour of manned mission activity, on the lunar surface in the period 1980-1990, and

ϕ ≡ Exponent of learning curve

The comparative cost of a candidate mobility system can now be expressed as

$\$_j$ ≡ cost of S_j as a mobility system

$$= \left[\left(\sum_{i=1}^{n} c_{ij} N_i\right) + m_j D_j\right] P + \left(\sum_{i=1}^{n} h_{ij} N_i\right) R + \left(\sum_{k=1}^{q} m_{jk} \$_{jk}\right) \left(D_j^{\phi} + B_j\right) \quad (9.4\text{-}1)$$

where

$\sum_{k=1}^{q} m_{jk} \$_{jk}$ first unit cost of S_j

$\sum_{i=1}^{n} h_{ij} N_i$ total hours of manned mission activity necessary for S_j to accomplish the specified missions to the same level as its effectiveness.

$\sum_{i=1}^{n} c_{ij} N_i$ total weight of consumables (food, fuels, O_2, etc.) needed by S_j to accomplish the specified missions to the same level as its effectiveness.

$\dot{m}_j D_j$ total dry weight of mobility systems to be developed on the lunar surface in the 1980-1990 period.

This equation for cost includes each of the cost factors discussed above. The term corresponding to each of these factors is shown in Table 9.4-1. The expression for first unit cost, F_j, i.e.

$$F_j = \sum_{k=1}^{q} m_{jk} \$_{jk} \qquad (9.4.2)$$

was developed based on the conclusion that compiling hardware item costs at the subsystem level would give the most promising cost estimate for first unit cost. This approach to first unit cost estimate will be developed further in Chapter 20. As shown in Table 9.4-1, the first unit cost can be used in the calculation of RDT&E cost, X_j, i.e.,

$$X_j = (\sum_{k=1}^{q} m_{jk} \$_{jk}) \quad B_j = F_j B_j$$

Cost estimation of the RDT&E cost can also be developed in terms of the hardware item RDT&E cost at this subsystem level. The development of such a procedure is treated in Chapter 20 also. In Chapter 20, careful consideration was given to the selection and definition of an appropriate set of hardware subsystems since not only knowledge of the subsystem weight must be available, but also considerable cost data should exist for each subsystem in order that reliable values of cost per pound can be determined.

A simplified form of Equation (9.4-1) was developed for use in this study. Defining

$$c_{ij} \equiv K (h_{ij}) + m_{ij}$$

where

K = pounds of crew consumables used per unit of crew time

h_{ij} = crew time used by S_j in performing M_i

TABLE 9.4-1. EXPRESSIONS FOR COST FACTORS

COST FACTOR	CORRESPONDING TERM
Research, Development, Testing & Evaluation Cost	$(\sum_{k=1}^{q} m_{jk}\ \$_{jk})\ B_j$
Production Cost	$(\sum_{k=1}^{q} m_{jk}\ \$_{jk})\ D_j^{\phi}$
Deployment Cost	$(m_j\ D_j)\ P$
Manned Mission Activity Cost	$(\sum_{i=1}^{n} h_{ij}\ N_i)\ R$
Cost of Consumables	$(\sum_{i=1}^{n} C_{ij}\ N_i)\ P$

m_{ij} = pounds of locomotion consumables used by S_j in performing M_i

and further defining

$h_{ij} \equiv T_{ij} \times \ell_i$, and

$m_{ij} \equiv R_{ij} \times f_j$,

where

T_{ij} = average crew time per member used by S_j in performing M_i,

ℓ_i = number of crew members required on M_i,

R_{ij} = range traversed by S_j in performing M_i, and

f_j = pounds of locomotion consumables used by S_j per unit of distance traversed.

Thus, the cost equation can now be written as

$$\$_j = \sum_{i=1}^{n} (T_{ij}\, l_i\, N_i) \quad (KP + R) + \sum_{i=1}^{n} (R_{ij}\, N_i)(f_j)(P) + m_j\, D_j\, P + F_j\, D_j^{\phi} + X_j \qquad (9.4\text{-}3)$$

With this form of the cost equation, the parameters necessary to calculate the comparative cost for any mobility system are easily identified. Of these parameters, T_{ij}, R_{ij}, f_j, m_j, F_j, and X_j are certainly critical system design parameters which directly reflect the required resource impact for any proposed mobility systems.

It is now possible to determine, using either Equation (9.4-1) or Equation (9.4-3), "how much" (i.e., the cost resource expenditure will be involved for each candidate mobility system. As with the effectiveness model, the sensitivity of the cost of a candidate system to changes in the values of the cost parameters can be evaluated.

9.5. Evaluation of Particular Systems

The evaluation methodology formulated above was initially employed to evaluate five proposed mobility systems. These systems were designed using mobility vehicles of various classes as the basic configuration and upgrading the systems where necessary to meet certain specifications of the gross system requirements of Chapter 5. For example, the basic lunar flyer is one-manned, while the defined guidelines specify that a buddy situation is desirable for extended traverse missions. Thus, the proposed candidate mobility system was conceived to consist of two separate units. These proposed systems are listed in Table 9.5-1. Table 9.5-2 and Table 9.5-3 list system data that are used in the calculation of effectiveness and cost, respectively. The values shown for first unit and RDT&E cost estimates were obtained from published documents for similar systems or interpreted from available vehicle information.

For the comparative cost calculations, values for ϕ, P, R, K, and ℓ were established as

ϕ = 1.0

P = \$1500 per pound

R = \$100,000 per hour of manned activity

TABLE 9.5-1 PROPOSED MOBILITY SYSTEMS

SYSTEM	BRIEF DESCRIPTION
Hopper	Pogo-stick effect, spring loaded, ballistic trajectory, one unit
Lunar Flyer	Rocket engine propulsion, reaction control, ballistic trajectory, two units
Lunar GEM	Plenum, differential pressure for elevation, gas jet control, two units
Manned/Remote Rover	Wheeled vehicle, electrical powered, unmanned remote operation, two units
Cabin Class Rover	Wheeled vehicle, habitable for extended traverse, one unit

TABLE 9.5-2. SYSTEM DESIGN PARAMETERS FOR EVALUATING EFFECTIVENESS

System	MISSION PARAMETERS Range (KM)		Payload* (LB)		Moving Time (HR)	
	Manned	Unmanned	Manned	Unmanned	Manned	Unmanned
Hopper	525	0	322	0	13	0
Lunar Flyer	24	0	600	0	0.1	0
Lunar GEM	40	0	400	0	1.5	0
Manned/Remote Rover	120	1000	400	300	39	500
Cabin Class Rover	400	0	1150	0	70	0

*Excluding Crew

TABLE 9.5-3. SYSTEM DESIGN PARAMETERS FOR EVALUATING COMPARATIVE COST

System	Dry Weight (lb)	f_j Fuel Rate (lb/km)	First Unit Cost (Millions)	RDT&E Cost (Millions)
Hopper	6500	10	28.2	423.6
Lunar Flyer	600	60	2.8	50.4
Lunar GEM	500	4	2.1	32.8
Manned/Remote Rover	1300	1	4.5	61.9
Cabin Class Rover	8100	3	27.3	415.3

$$K = 2.0$$

$$\ell_i = \begin{cases} 0, & \text{unmanned mission} \\ 2, & \text{manned mission} \end{cases}$$

Further, the number of each mission to be performed and the number of mobility vehicles required were obtained from an analysis of the projected integrated program plan schedule of tug landings and lunar missions presented in Chapter 5 of this report. It was determined that fifteen mobility systems would be required during the 1980-1990 period, i.e. $D_j = 15$. A listing of the projected number of each mission is given in Table 9.5-4.

Before the effectiveness model could be used to calculate the effectiveness of any candidate, it was necessary to assign numerical values to the mission weights, W_i. The values selected are shown in Table 9.5-5.

Although the calculations necessary to obtain the effectiveness and cost for each alternative system can be performed by tabular methods without difficulty, a digital computer program was prepared to compute effectiveness and cost. (See Appendix E). This program offers significant opportunity to perform detailed sensitivity studies of the mobility system performance parameters.

As an example of the procedure used to calculate a value for the adequacy of a candidate to perform a particular mission, consider the ability of the Hopper to perform mission M_3. Using Equation (9.3-1) in conjunction with data from Table (9.3-1) and Table (9.5-2), then,

TABLE 9.5-4.

PROJECTED NUMBER OF MISSIONS

Mission	NUMBER OF MISSIONS
M_1	50
M_2	50
M_3	192
M_4	192
M_5	190
M_6	18
M_7	7
M_8	18
M_9	19
M_{10}	18

TABLE 9.5-5

MISSION WEIGHT VALUES

Mission	Weight Value
1	0.10
2	0.10
3	0.20
4	0.25
5	0.20
6	0.03
7	0.03
8	0.03
9	0.03
10	0.03

$$A_{31} = \frac{1}{3}\left[\frac{13}{26} + \frac{525}{250} + \frac{322}{500}\right]$$

Thus,

$$A_{31} = \frac{1}{3}(.50 + 1.0 + .64) = 0.71$$

Note that the second term on the right hand side was made equal to 1.0 in the actual calculation of A_{31}, in accord with the effectiveness model. From a Hopper design point of view, this may indicate that a tradeoff between range and, say, payload should be considered. Of course, the Hopper's performance for the longer missions must also be considered before such a decision is finalized.

From the considerations necessary to determine a system's adequacy for a given mission, it is convenient to also be tabulating the range and crew time for the mission. Tables 9.5-6 to 9.5-10 present the results of the adequacy calculations, etc., for the proposed mobility systems. Calculation of the effectiveness of the proposed systems is now straightforward. The calculations in computing the cost estimates are also straightforward, but longer. The calculated values of effectiveness and comparative cost are shown in Table 9.5-11 and plotted on the effectiveness cost graph of Figure 9.5-1. An established criterion would now be used to rank these alternatives in order of desirability to assist in the task of selecting the most promising mobility system for the 1980-1990 period. It should be remembered that only the manned/remote rover alternative could perform the unmanned mission activities specified by the gross system requirements.

TABLE 9.5-6 CALCULATED ADEQUACIES: HOPPER

Mission	A_{ij}	R_{ij} (km)	T_{ij} (hr)
M_1	0	0	0
M_2	0	0	0
M_3	0.71	250	13
M_4	0.59	250	13
M_5	1.00	5	4
M_6	0	0	0
M_7	0	0	0
M_8	0.70	6	1
M_9	0	0	0
M_{10}	0.70	6	1

TABLE 9.5-7. CALCULATED ADEQUACIES: 1 LUNAR FLYER

Mission	A_{ij}	R_{ij} (km)	T_{ij} (hr)
M_1	0	0	0
M_2	0	0	0
M_3	0.37	24	0.1
M_4	0.21	24	0.1
M_5	1.0	5	0.1
M_6	0	0	0
M_7	0	0	0
M_8	0.77	6	0.1
M_9	0	0	0.1
M_{10}	0.77	6	0.1

TABLE 9.5-8

CALCULATED ADEQUACIES: LUNAR GEM

Mission	A_{ij}	R_{ij} (km)	T_{ij} (hr)
M_1	0	0	0
M_2	0	0	0
M_3	0.34	40	2
M_4	0.19	40	2
M_5	1.0	5	1
M_6	0	0	0
M_7	0	0	0
M_8	0.73	6	1
M_9	0	0	0
M_{10}	0.73	6	1

TABLE 9.5-9

CALCULATED ADEQUACIES: MANNED/REMOTE ROVER

Mission	A_{ij}	R_{ij} (km)	T_{ij} (hr)
M_1	0.57	1000	500
M_2	0.48	1000	500
M_3	0.76	120	26
M_4	0.36	120	16
M_5	1.0	5	4
M_6	1.0	10	12
M_7	1.0	300	150
M_8	0.75	6	1
M_9	0.75	6	6
M_{10}	0.75	6	1

TABLE 9.5-10

CALCULATED ADEQUACIES: CABIN CLASS ROVER

Mission	A_{ij}	R_{ij} (km)	T_{ij} (hr)
M_1	0	0	0
M_2	0	0	0
M_3	1.0	250	26
M_4	0.63	160	16
M_5	1.0	5	4
M_6	1.0	10	12
M_7	0	0	0
M_8	0.86	6	1
M_9	0	0	0
M_{10}	0.86	6	1

TABLE 9.5-11

SUMMARY OF EFFECTIVENESS COST CALCULATIONS

System	Effectiveness	Cost (Millions)
Hopper	0.57	3,341
Lunar Flyer	0.37	1,035
Lunar GEM	0.42	335
Manned/Remote Rover	0.68	2,083
Cabin Class Rover	0.64	2,522

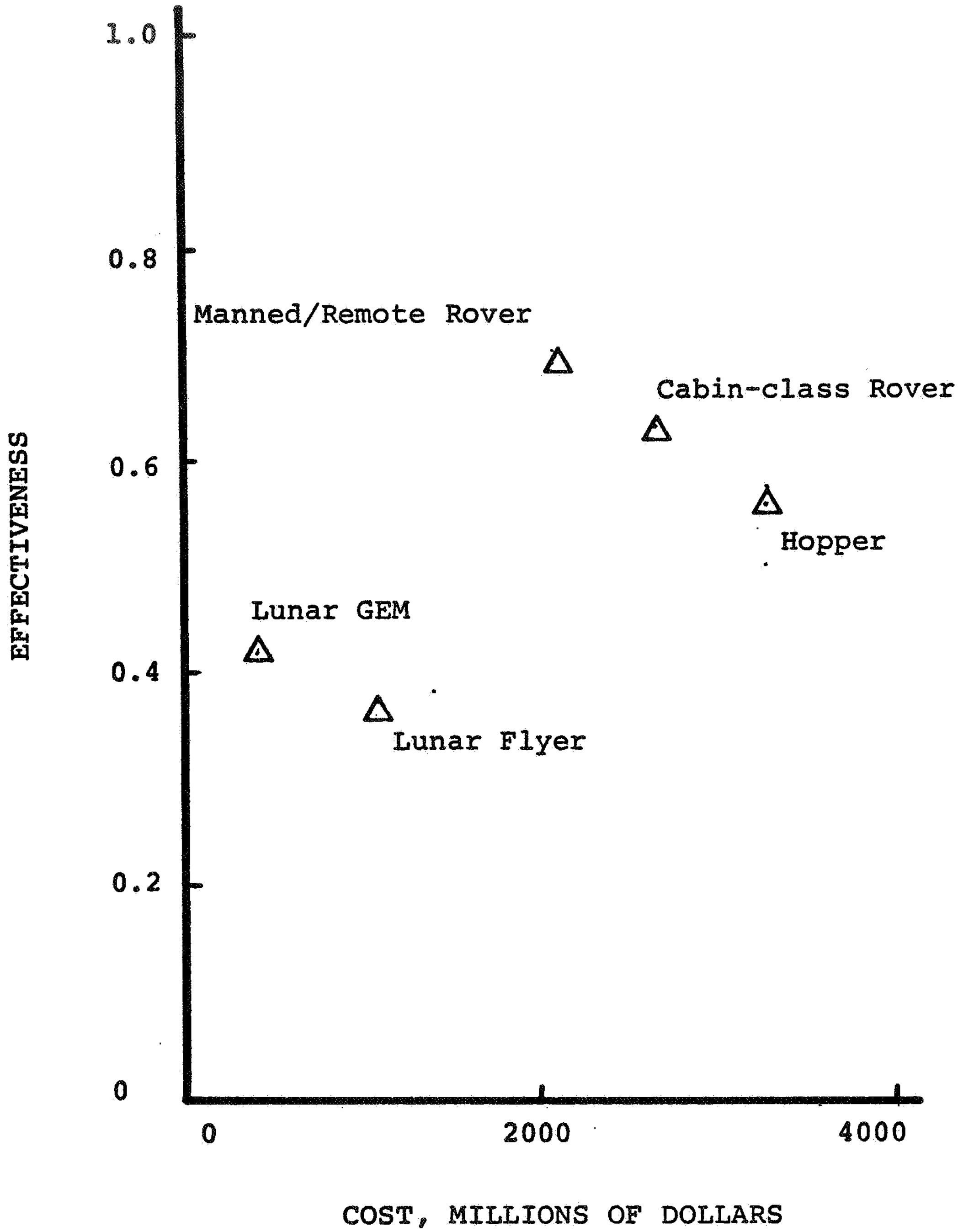

FIGURE 9.5-1 EFFECTIVENESS COST FOR PROPOSED CANDIDATE MOBILITY SYSTEMS

TABLE 9.5-12 COMPARISON OF COST FACTORS FOR PROPOSED SYSTEMS

System	COMPARATIVE COST FACTORS				
	RDT&E ($X10^6$)	Production ($X10^6$)	Deployment ($X10^6$)	Mantime ($X10^6$)	Consumables ($X10^6$)
Hopper	\$423.6	\$423.8	\$146.3	\$863.2	\$1,485.0
Lunar Flyer	50.4	42.0	13.5	12.2	917.0
Lunar GEM	32.8	31.5	11.3	198.8	60.3
Manned/ Remote Rover	61.9	67.5	29.1	1,813.2	111.6
Cabin Class Rover	415.3	409.5	182.3	1,159.2	356.0

The rover class of mobility vehicle, based on the results of these proposed systems, appeared to be the most promising class of mobility systems. However, since the proposed candidates were not considered to represent the optimum design of a particular class of mobility systems, numerous sensitivity studies should be performed on each class of mobility system to study its behavior on the effectiveness cost graph.

As an example of such a sensitivity study, a modification of the lunar GEM with the system upgraded to enable it to travel 120 km manned and equipped for remote operation was considered. The vehicle parameters then were as follows:

Range:	120 km manned, 40 km unmanned
Payload:	400 lb manned, 800 lb unmanned
Moving time:	1.5 hr manned, 8 hr unmanned

Calculation of the corresponding effectiveness showed an encouraging ten percent increase to 0.52. Aided by the prepared computer program, detailed sensitivity studies are indeed realistic. It would be possible, for example, to study a system's cost sensitivity to improvements in the vehicle's fuel consumption rate or the cost penalty involved for imposing an accelerated development in technology. Another desirable study would be the sensitivity of the effectiveness of a system to variations in its unmanned operations.

The magnitude of the various cost factors of the comparative cost were computed. Table 9.5-12 provides a comparison of these values

TABLE 9.5-13. COMPARISON OF REQUIRED CONSUMABLES FOR PROPOSED SYSTEMS

System	Mantime Involved (10^3HR)	CONSUMABLES Crew (10^3 LB)	CONSUMABLES Locomotion (10^3LB)	CONSUMABLES Total (10^3LB)
Hopper	8.6	23.0	968.0	991.0
Lunar Flyer	0.1	0.2	611.0	611.2
Lunar GEM	2.0	4.0	36.2	40.2
Manned/Remote Rover	18.1	37.0	37.4	74.4
Cabin Class Rover	11.6	37.0	200.0	237.0

for the five proposed systems. Comparison of the systems in terms of manned mission activity time and necessary consumables is shown in Table 9.5-13.

9.6. Potential Iterations

The key to successful development of a methodology as presented above is iteration - a continuous cycle of selecting the objectives, designing the criterion, formulating the models, collecting data, proposing alternatives, weighting effectiveness against cost, questioning assumptions, reexamining the models, and so on until satisfaction is obtained.

One proposed iteration of the effectiveness model, for example, would be to expand the formulation to include the following consideration:

- o Payload - Volume, weight, peak and total power requirements, data transmission bandwidth.
- o Traverse - Length, maximum and minimum time, number of stops, moving time required, mantime to operate experiments, mantime to operate vehicle, EVA time required.
- o Reliability - Sensitivity of system to short term failure of locomotion subsystem and power subsystem, rough lunar surface, impaired motion by obstacle, loss of communication with remote control base, loss of automatic guidance.
- o Human Factors - Crew comfort, ease of control, crew training time, ease of ingress/egress, time to ingress/egress, thermal control, surface visibility, simplicity of vehicle repair.

In terms of the prime objective of choosing the most desirable alternative for a class of mobility system for the future, valuable insight might also be obtained by using the developed methodology to examine combinations of proposed mobility systems. It is further suggested that studies be conducted using the fixed budget and the specified effectiveness approach where one of the measures of the alternative systems' design, e.g., cost, is pre-established and the candidates evaluated in terms of their other measure of system design.

9.7. Evaluation of Synthesized Mobility System

In Chapter 20 of this report, a methodology for determining the first unit and RDT&E cost estimates is formulated. The procedure developed is presented using the synthesized mobility system as an example. The totals for the cost estimates of the first unit and RDT&E for the synthesized vehicle are $26 million and $458 million, respectively.

The mobility system synthesis committee had as its objective the synthesis of a mobility system that could perform all the specified missions of the gross system requirements. Thus, the synthesized vehicle had an effectiveness very close to 1. The comparative cost for the synthesized system, calculated using the cost model, was $3,870 million. A breakdown of this cost into the various cost factors is shown in Table 9.7-1. In Figure 9.7-1 the synthesized system is plotted, along with the five proposed mobility systems for comparison.

TABLE 9.7-1. COMPARATIVE COST SUMMARY FOR SYNTHESIZED VEHICLE COST IN MILLIONS OF 1970 DOLLARS

Research, Development, Testing, and Evaluation	458
Vehicle Production	402
Lunar Deployment	203
Required Consumables	991
Manned Mission Activity	1,816
	3,870

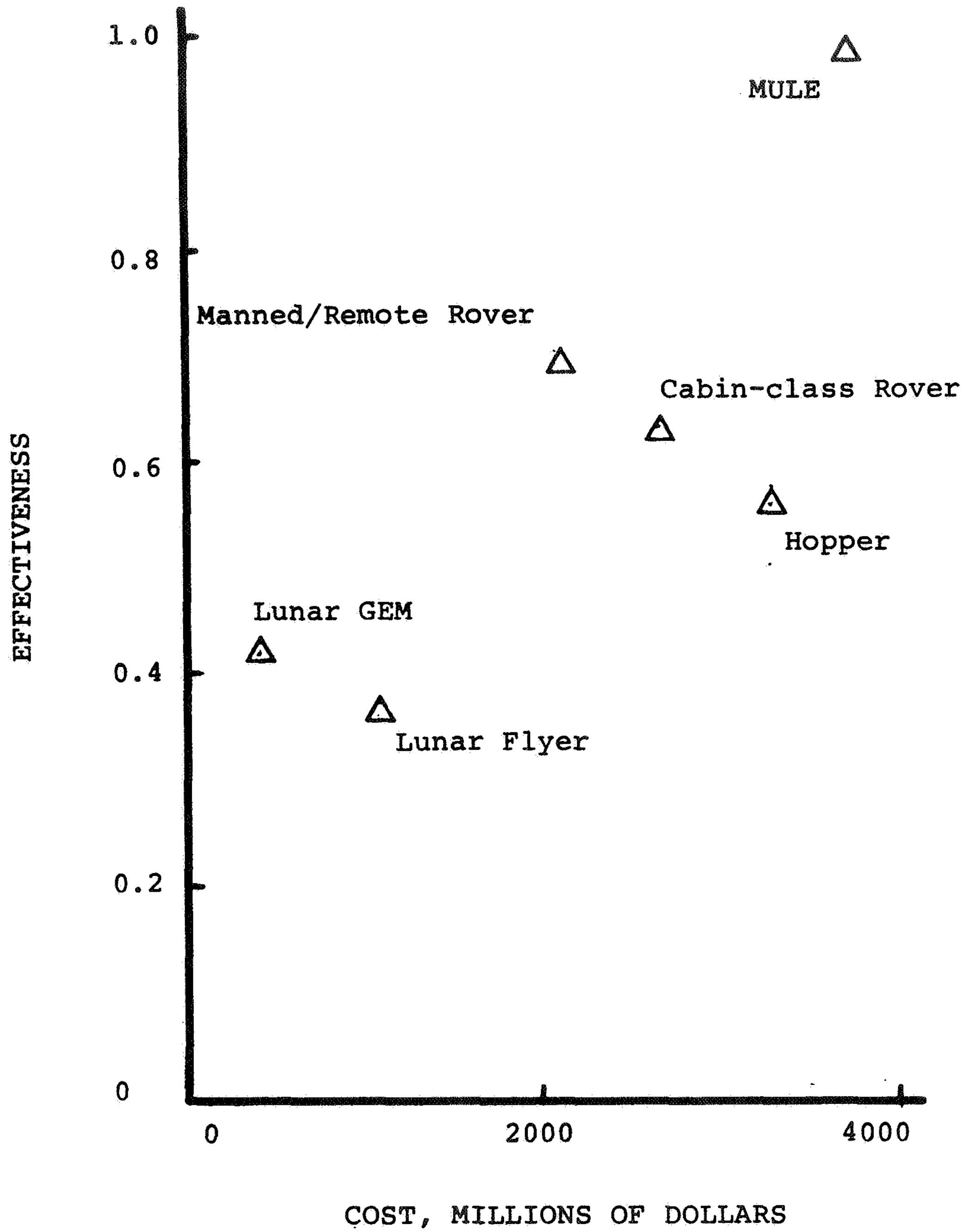

FIGURE 9.7-1 EFFECTIVENESS COST FOR SYNTHESIZED MOBILITY SYSTEM

Within the framework of the nature and scope of this study, it can be concluded that the synthesized mobility system is considerably more effective in performing lunar surface activities in the 1980-1990 period than the other alternatives evaluated for only a slight incremental increase in comparative cost.

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CHAPTER 10

SYNTHESIS

Leo R. Pucacco

10.1. Introduction

The recommendation made at the end of Phase I was that a near ground mobility vehicle, known as the MULE, be designed and evaluated. However, the designation near ground encompasses a large number of possible configurations. For example, consider the best class of vehicles from Phase I, the rover class. The question can be asked, what is a rover? Does it have six wheels like Molab or four legs like a quadraped? Both of these plus many other configurations from Phase I were defined as rover classes. The ground effects machine (GEM) was sighted in Phase I as a near ground system that could be the configuration of MULE. Is the Lunagem the best GEM configuration? How many plenums should a GEM have and is the reaction jet the best means of propulsion?

Phase I eliminated the hopper and flyer class of vehicles and recommended near ground system. Therefore, the design and evaluation of a near ground MULE was left to Phase II. In order to accomplish this task systems engineering was utilized. First, the development of as many near ground systems as possible had to be accomplished. These systems would then be evaluated and the final system design would evolve. This chapter deals with the procedure followed and results obtained from the development of a near ground MULE.

10.2. Development of Synthesis Concept

The process by which a final system design evolves in systems engineering is illustrated in Figure 10.2-1.

Another process by which a final system design can be determined is through subsystems synthesis. The subsystem synthesis technique is illustrated in Figure 10.2-2.

The locomotion subsystem was the only one which was considered in the synthesis procedure for the following reasons:

1. Preliminary parametric studies indicated that the weight associated with the locomotion function is the predominant design factor.
2. The entire design of the MULE is based upon the problem of dynamic and static contact with the lunar terrain.
3. The astrionics subsystem is basically unique and unaffected by other subsystems.
4. The power subsystem is affected directly by locomotion and is linked to locomotion.
5. Time limitations allowed only a first order evaluation of the subsystems.

The locomotion subsystem is the one which interacts with the surface to cause motion and suspends the cabin, payload and subsystems supporting them.

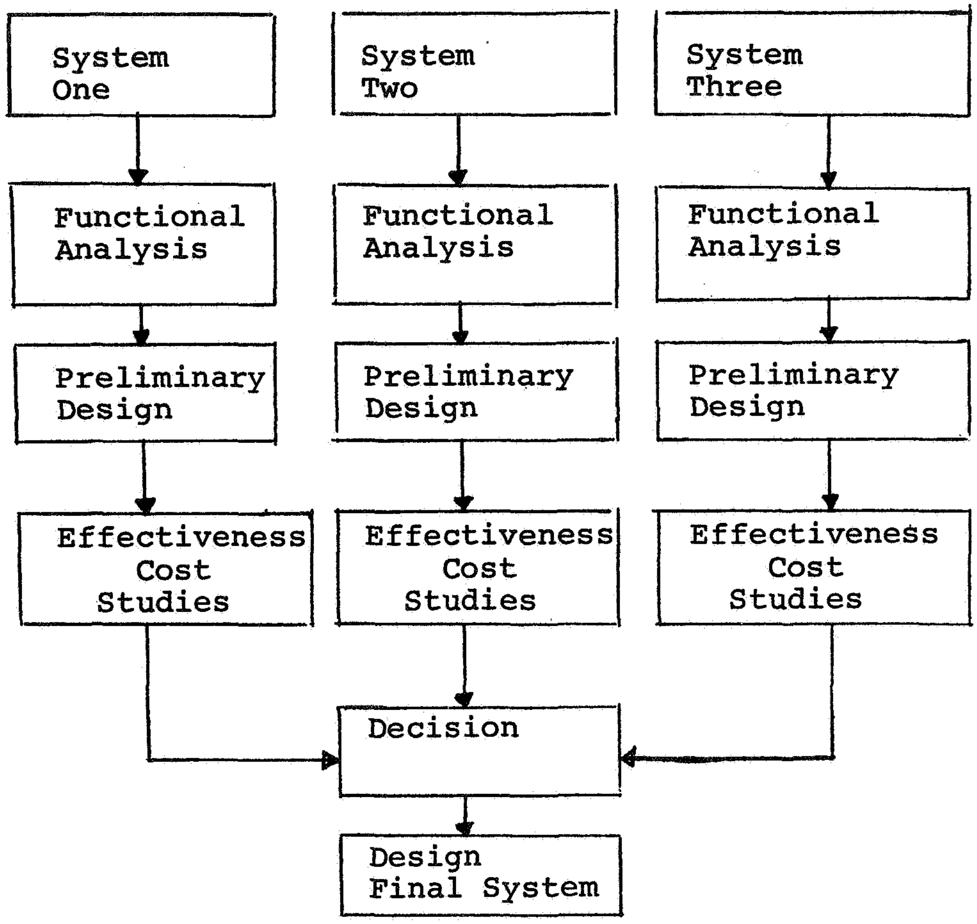


FIGURE 10.2-1. SYSTEMS ENGINEERING PROCESS

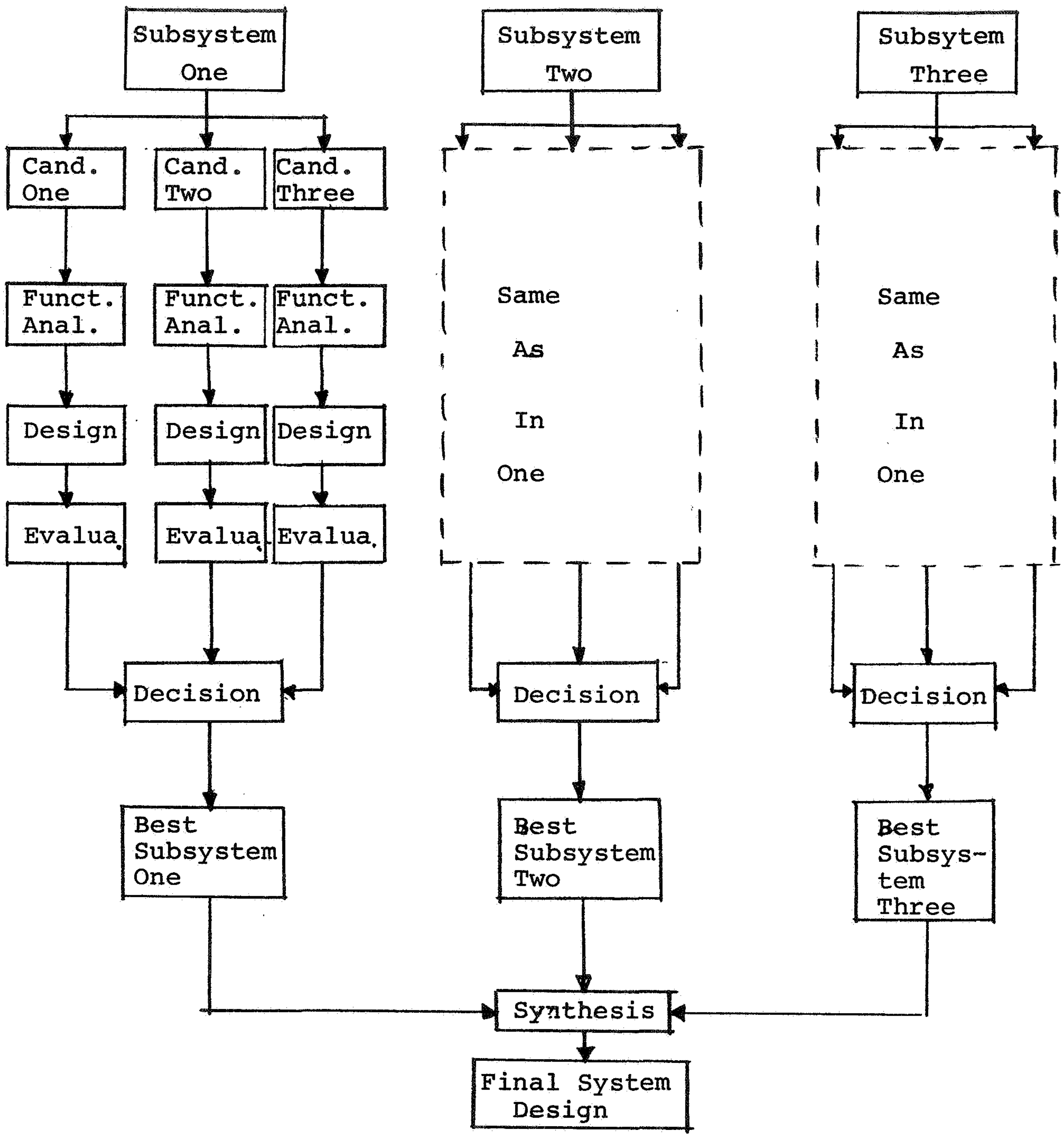


FIGURE 10.2-2. SUBSYSTEMS SYNTHESIS

10.3. Subsystems Synthesis of the MULE

In the case of the MULE there are four subsystems:

1. LOCOMOTION
   a. STRUCTURE
   b. MOBILE CONTROL
   c. PROPULSION
2. CABIN
   a. ENVIRONMENTAL CONTROL
   b. LIFE SUPPORT
3. ASTRIONICS
   a. COMMUNICATIONS
   b. NAVIGATION & GUIDANCE
   c. HAZARD DETECTION & AVOIDANCE
   d. REMOTE CONTROL & TELEOPERATORS
   e. COMPUTER
4. POWER
   a. RTG
   b. FUEL CELL
   c. BATTERIES
   d. POWER MANAGEMENT

The subsystem synthesis procedure for the MULE is illustrated in Figure 10.3-1.

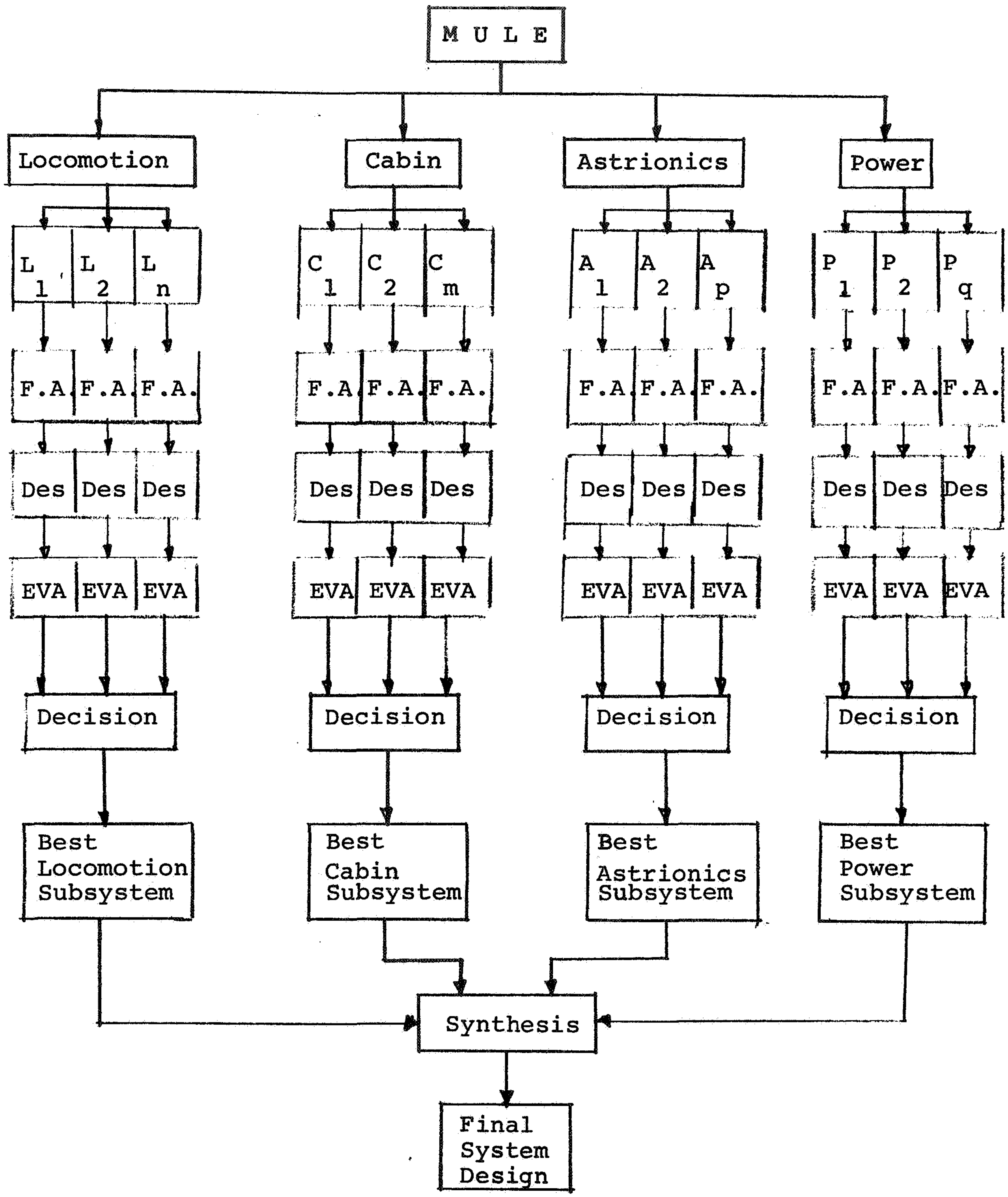


FIGURE 10.3-1. SUBSYSTEMS SYNTHESIS OF MULE

10.4. Candidate Locomotion Subsystems

A preliminary list of locomotion subsystems which are near ground in nature are as follows:

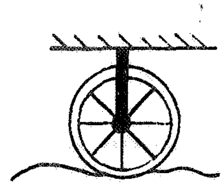
1. Wheel
2. Rimless Wheel
3. Ski
4. Track
5. Screw (Auger)
6. Reaction Jet
7. Dust Jet
8. Rod (Shaft)
9. Jointed Leg
10. Ball & Socket
11. Vibrating Leg
12. Hydraulic Leg

The list was studied carefully in an effort to consolidate basically similar candidates and add any new ones which might have been overlooked. The rimmed and unrimmed wheel were combined and considered as a single candidate. The behavior of the dust jet was not sufficiently different from the conventional hot-gas jet to consider it separately. The hydraulic leg was considered a special case of the jointed leg and, therefore, was combined with it. The vibrating leg was eliminated because of ambiguity of concept and relative complexity. Therefore, the number of candidate locomotion subsystems was reduced to nine and are shown in Figure 10.4-1.

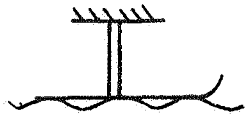
10.5. Functional Analysis of Locomotion

The first step taken after the candidate locomotion subsystems were

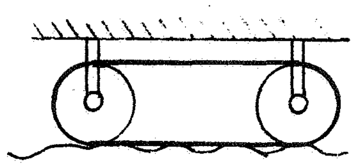
I Wheel (rimmed or unrimmed)



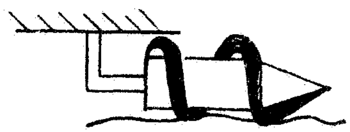
II Ski



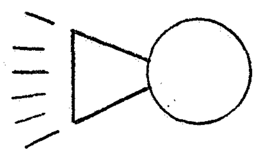
III Track



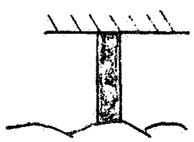
IV Auger



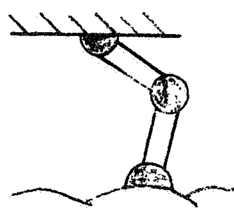
V Reaction Jet



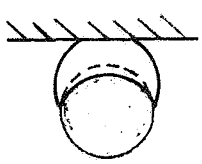
VI Shaft



VII Jointed Leg



VIII Ball & Socket



IX Plenum

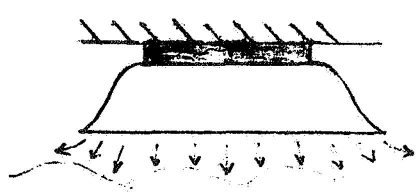


FIGURE 10.4-1. LOCOMOTION SUBSYSTEM CANDIDATES

10.5. Functional Analysis of Locomotion

defined was to perform a functional analysis. The functional analysis was utilized to:

(1) Document the requirements of the locomotion subsystem.

(2) Document the means by which a particular locomotion candidate will perform the required tasks.

(3) To find candidates which will be unable to perform the required tasks.

(4) To infer the complexity of control, structure and additional subsystems which will be required by a given candidate.

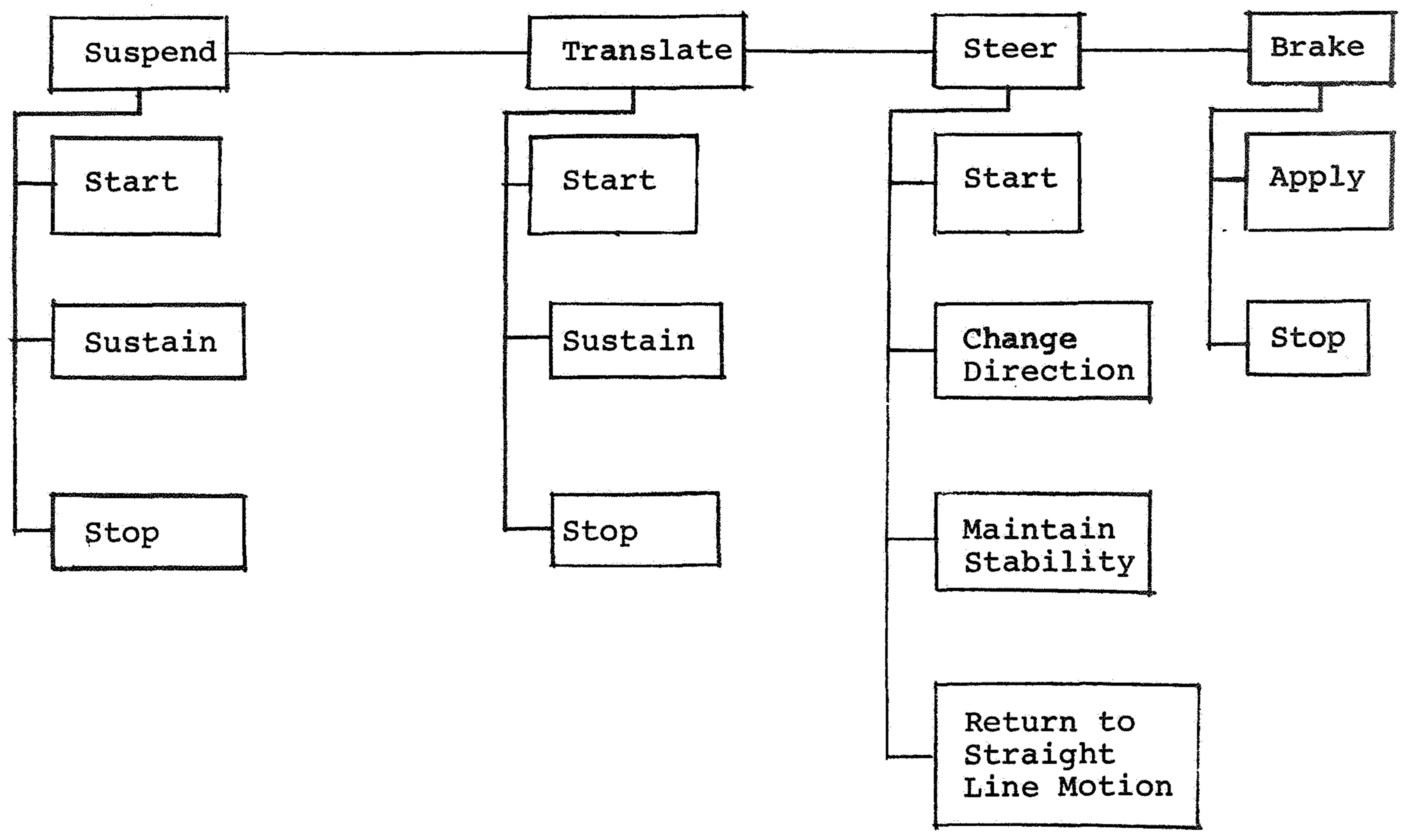
(5) To identify the interfaces necessary with the remainder of the subsystems.

The first and second level functional analysis for locomotion is presented in Figure 10.5-1.

The functional analysis provided the four primary functional requirements of locomotion: (1) Suspending, (2) Translating, (3) Steering and (4) Braking.

10.6. Evaluation of Locomotion Subsystems

In order to evaluate the nine candidate locomotion subsystems a criteria was necessary. It was decided to determine the effectiveness of each candidate in accomplishing the requirements of locomotion. To do this, indices of adequacy were developed which measured the effectiveness of the locomotion subsystem. The indices of adequacy were based on: (1) statement of work, (2) mission



FIRST AND SECOND LEVEL FUNCTIONAL ANALYSIS FOR LOCOMOTION
FIGURE 10.5-1

requirements (3) functional analysis of locomotion and (4) design team guidelines. Table 10.6-1 presents the inputs utilized in developing the indices of adequacy.

	Input	Source
1.	Payload weight at least 2000 pounds	Statement of work
2.	Maintain level platform	Design Team
3.	Dampen Vibration	Design Team
4.	Ground clearance: 50 centimeters	Statement of Work
5.	Maintain position on 45° sideslope	Statement of Work
6.	Speed of 10 kilometers per hour required	Mission Requirements
7.	Climb 30° slope	Statement of Work
8.	Negotiate 90 centimeter crevasse	Statement of Work
9.	Negotiage 90 centimeter step	Statement of Work
10.	Provide bulldozing capability	Mission Requirements
11.	Have low resistance to motion	Design Team
12.	Mechanical simplicity in steering	Design Team
13.	Energy for steering minimal	Design Team
14.	Responsive steering	Design Team
15.	Short stopping distance	Design Team
16.	Stability during stopping	Design Team
17.	Energy for braking minimal	Design Team
18.	Descend 30° slope	Statement of Work
19.	First level functional analysis of locomotion	Systems Engineering

TABLE 10.6-1 SOURCES FOR INDICES OF ADEQUACY

From the information presented in Table 10.6-1 seven indices of adequacy were developed. In addition, each index was assigned a weighting factor in the form of locomotion units. These locomotion units were distributed in sub-sections of each index. The total number of locomotion units is 120.

INDICES OF ADEQUACY

I.	PERFORMANCE WHILE SUSPENDING	
	A. Payload/Gross weight	8 units
	B. Stability	8 units
	1. Level terrain	
	2. On 45° slope	
	C. Vibration damping properties	4 units
II.	PERFORMANCE WHILE TRANSLATING	
	A. Speed	7 units
	B. Push-pull capability	8 units
	1. Climb 30° slope	
	2. Negotiate 90 centimeter crevasse	
	3. Negotiate 50 centimeter step	
	4. Bulldozing capability	
	C. Resistance to motion	5 units
III.	PERFORMANCE WHILE STEERING	
	A. Mechanical simplicity	6 units
	B. Energy required	4 units
	C. Controllability	10 units
	1. Turn radius	
	2. Response	
	3. Overshoot	
IV.	PERFORMANCE WHILE BRAKING	
	A. Stopping distance	8 units
	1. Soft surface	
	2. Hard surface	
	B. Stability	8 units
	C. Energy required	4 units
V.	PROVISION OF TRANSLATION	
	A. Speed	7 units
	B. Push-pull capability	8 units
	C. Resistance to motion	5 units
VI.	PROVISION OF STEERING	
	A. Mechanical simplicity	3 units
	B. Energy required	2 units
	C. Controllability	5 units

VII. PROVISION OF BRAKING

A. Stopping distance — 4 units
B. Stability — 4 units
C. Energy required — 2 units

10.7 RESULTS OF EVALUATION: In utilizing the indices of adequacy to evaluate the nine locomotion subsystem candidates a comparison technique was employed. The maximum number of locomotion units available for each adequacy index subdivision was assigned to the best of the nine candidates. The other candidates were then rated on a comparison basis.

PERFORMANCE WHILE SUSPENDING

20 Locomotion Units

Candidate	A	B	C	Total
Wheel	5	6	3	14
Ski	7	5	2	14
Track	3	8	2	13
Auger	3	8	1	12
Reaction Jet	0	0	0	0
Shaft	5	4	0	9
Jointed Leg	3	4	2	9
Ball & Socket	6	5	3	14
Plenum	8	5	4	17

The results of the evaluation of the nine locomotion subsystems are summarized in Table 10.7-1.

PROVISION OF STEERING

10 Locomotion Units

Candidate	A	B	C	Total
Wheel	3	1	4	8
Ski	3	1	2	6
Track	3	1	5	9
Auger	3	1	4	8
Reaction Jet	3	1	4	8
Shaft	1	1	3	5
Jointed Leg	1	1	3	5
Ball & Socket	0	0	0	0
Plenum	0	0	0	0

PROVISION OF BRAKING

10 Locomotion Units

Candidate	A	B	C	Total
Wheel	2	3	1	6
Ski	1	2	0	3
Track	4	3	2	9
Auger	4	3	2	9
Reaction Jet	4	4	0	8
Shaft	4	2	1	7
Jointed Leg	3	2	1	6
Ball & Socket	1	2	1	4
Plenum	0	0	0	0

PERFORMANCE WHILE BRAKING

20 Locomotion Units

Candidate	A	B	C	Total
Wheel	5	6	2	13
Ski	3	3	0	6
Track	8	7	4	19
Auger	8	7	4	19
Reaction Jet	0	0	0	0
Shaft	8	4	3	15
Jointed Leg	6	4	2	12
Ball & Socket	3	5	1	9
Plenum	0	8	0	8

PROVISION OF TRANSLATION

20 Locomotion Units

Candidate	A	B	C	Total
Wheel	7	8	4	19
Ski	0	0	0	0
Track	7	8	4	19
Auger	4	4	1	9
Reaction Jet	7	5	2	14
Shaft	4	4	5	13
Jointed Leg	4	8	3	15
Ball & Socket	0	0	0	0
Plenum	0	0	0	0

PERFORMANCE WHILE TRANSLATING

20 Locomotion Units

Candidate	A	B	C	Total
Wheel	7	8	4	19
Ski	7	6	2	15
Track	7	8	4	19
Auger	4	4	1	9
Reaction Jet	0	0	0	0
Shaft	4	4	5	13
Jointed Leg	4	8	3	15
Ball & Socket	7	8	4	19
Plenum	7	8	5	20

PERFORMANCE WHILE STEERING

20 Locomotion Units

Candidate	A	B	C	Total
Wheel	6	3	8	17
Ski	6	2	5	13
Track	6	2	10	18
Auger	6	1	9	16
Reaction Jet	0	0	0	0
Shaft	3	2	5	10
Jointed Leg	3	2	5	10
Ball & Socket	6	0	2	8
Plenum	6	0	2	8

Candidate Locomotion Subsystems	I Performance While Suspending	II Performance While Translating	III Performance While Steering	IV Performance While Braking	V Provision of Translation	VI Provision of Steering	VII Provision of Braking	TOTAL LOCOMOTION UNITS
Wheel	14	19	17	13	19	8	6	96
Ski	14	15	13	6	0	6	3	70
Track	13	19	18	19	19	9	9	106
Auger	12	9	16	19	9	8	9	82
Reaction Jet	0	0	0	0	14	8	8	30
Shaft	9	15	10	12	15	5	6	72
Jointed-Leg	9	15	10	12	15	5	6	72
Ball & Socket	14	19	8	9	0	0	4	54
Plenum	17	20	8	8	0	0	0	53

TABLE 10.7-1 SUMMARY OF LOCOMOTION SUBSYSTEMS

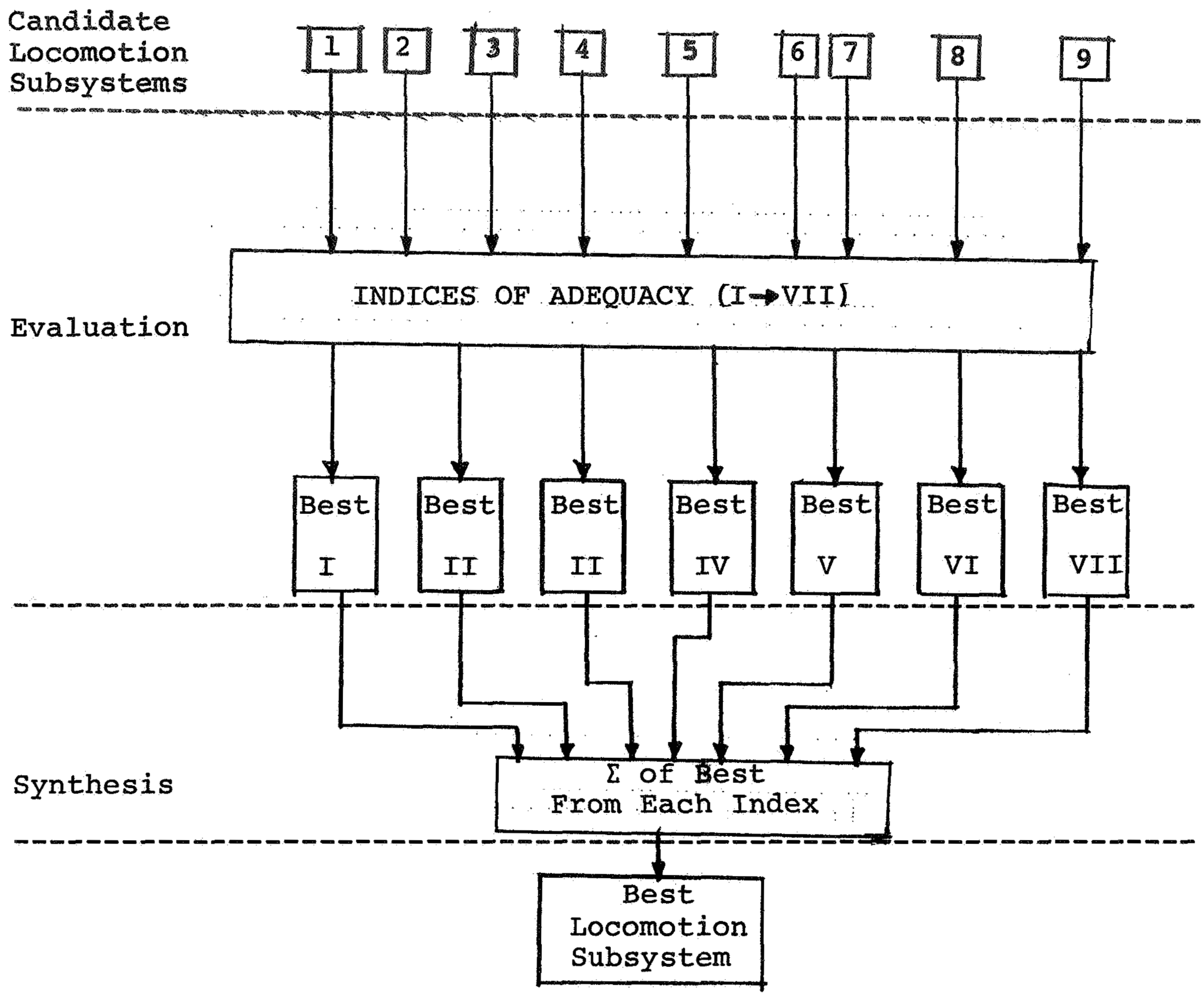
10.8 Synthesis of Prime Candidates

The nine candidate subsystems listed in order of total locomotion units along with their effectiveness are presented in Table 10.8-1.

Candidate	Locomotion Units	Effectiveness $\left(\frac{\text{Units}}{120}\right)$
Track	106	0.883
Wheel	96	0.800
Auger	82	0.683
Shaft	72	0.600
Jointed-Leg	72	0.600
Ski	70	0.583
Ball & Socket	54	0.450
Plenum	53	0.250

TABLE 10.8-1 RESULTS OF EVALUATION

As Table 10.8-1 indicates the most effective single concept (primary) locomotion subsystem is the track, followed by the wheel, auger, shaft and so on. At this point the process of synthesis as defined previously was modified to produce the most effective locomotion subsystem considering all concepts. That is, by combining two or more locomotion concepts into one, the most effective locomotion subsystem will result. This process is illustrated in Figure 10.8-1.



DEVELOPMENT OF MULTIPLE CONCEPT SUBSYSTEM
FIGURE 10.8-1

Utilizing the procedure outlined in Figure 10.8-1 and the results of Table 10.7-1 the best candidate for each of the seven indices of adequacy was determined and listed in Table 10.8-2.

I.	Performance while suspending	17	Plenum
II.	Performance while translating	20	Plenum
III.	Performance while steering	18	Track
IV.	Performance while braking	19	Track, Auger
V.	Provision of translation	19	Track, Wheel
VI.	Provision of steering	9	Track
VII.	Provision of braking	9	Track, Auger

TABLE 10.8-2 BEST MULTIPLE CONCEPT SUBSYSTEM

The results presented in Table 10.8-2 indicate that the best multiple concept locomotion subsystem, which will be denoted as a hybrid, is a combination Plenum - Track - Auger - Wheel. This hybrid will have an effectiveness of 111/120 = 0.925. At this point a decision was made on the auger. There was reason to believe that the auger would not be capable of attaining the speed required (at least 10 kilometers per hour). In addition, its draw-bar pull is substantially below the track and offers no advantage over the track. For these reasons the auger was eliminated as a single locomotion subsystem or part of a hybrid subsystem. It was further decided to limit the synthesis to two locomotion candidates to form a dual hybrid. The dual hybrids which ranked favorably are presented in Table 10.8-3.

Indices of Adequacy	Plenum-Track	Track-Wheel	Plenum-Wheel	Plenum-Reaction Jet
I	17	14	17	17
II	20	19	20	20
III	18	18	17	8
IV	19	19	13	8
V	19	19	19	14
VI	9	9	6	8
VII	9	9	6	8
TOTAL	111	107	100	83
Effectiveness	0.925	0.892	0.833	0.692

TABLE 10.8-3 DUAL HYBRID RESULTS

From the results of Tables 10.8-1 and 10.8-3 it is apparent that the dual hybrids do not provide a marked increase in effectiveness over some of the single concept candidates. In order to illustrate this point and provide a basis for a decision on the locomotion subsystem, Table 10.8-4 lists those candidates which have an effectiveness equal to or greater than 0.8.

Locomotion Subsystem	Total Units	Effectiveness
Plenum & Track	111	0.925
Track & Wheel	107	0.892
Track	106	0.883
Plenum & Wheel	100	0.833
Wheel	96	0.800

TABLE 10.8-4 TOP FIVE LOCOMOTION SUBSYSTEMS

The results of Table 10.8-4 indicate that synthesis did not determine the best locomotion subsystem because the differences between the top three candidates were not significant enough. What synthesis did accomplish, however, was to provide a list of locomotion subsystems, in order of effectiveness, with enough basis to allow a further decision to be confidently made. That is, we now had to decide between the top five or so candidates.

10.9 Conclusion of Subsystem Synthesis

The results of the subsystems synthesis procedure was that the most effective locomotion subsystem is the plenum and track hybrid. However, because the effectivenesses of the track and wheel and track subsystems were not sufficiently lower they deserved further consideration. It was decided to do a preliminary design on these three candidates in the hope that enough information could be acquired to eliminate two on a solid basis.

PART III

CONFIGURATION

Chapters 11 through 15 cover the conceptual designs of the final MULE system, along with each of the individual major subsystems. These subsystems are Locomotion, Cabin, Power and Astrionics.

CHAPTER 11

SYSTEM CONFIGURATION

Leo R. Pucacco

11.1. Introduction

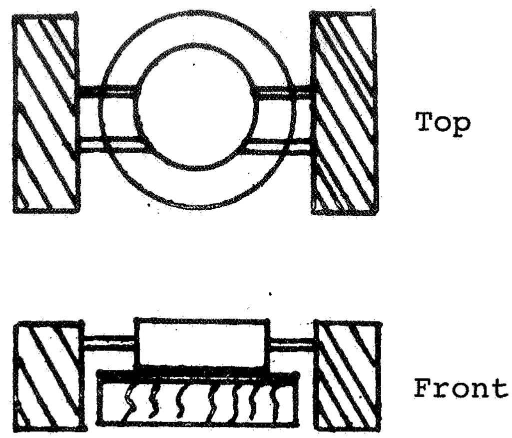
The result of subsystem synthesis was that based upon the locomotion subsystem a preliminary design of three configurations should be made. The result of this preliminary design would then provide a basis for the selection of a locomotion subsystem for the final configuration of the MULE.

In order to accomplish this task it was necessary to decide on one configuration for each of the three locomotion subsystem candidates. The three candidates which evolved from subsystem synthesis were: (1) plenum with track, (2) track, and (3) track with wheel. The decision on one of these three locomotion subsystems would provide the basis for the final system configuration. Incorporating the other three subsystems, astrionics, cabin and power, the final configuration of the MULE would evolve.

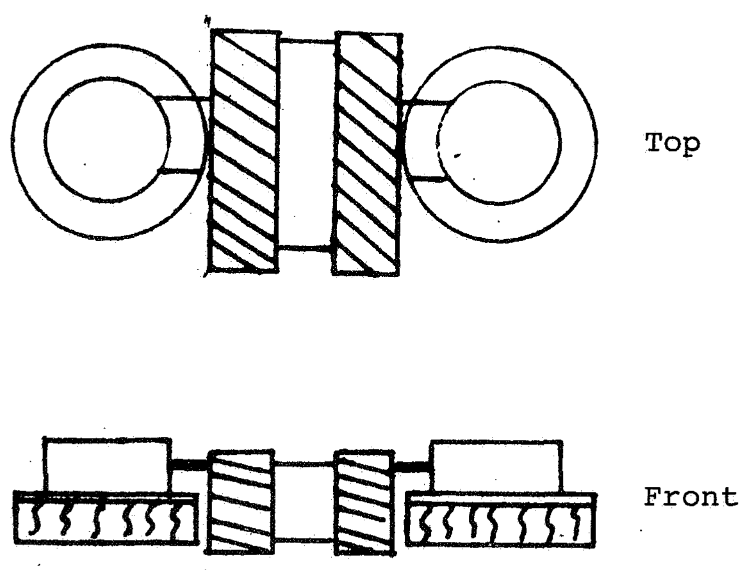
11.2. Configurations of the Three Viable Locomotion Subsystems

11.2.1 Plenum and Track

The first locomotion subsystem candidate considered was the plenum and track. The configurations incorporating both the plenum and track concept which were considered are illustrated in Figures 11.2-1, 11.2-2, and 11.2-3.

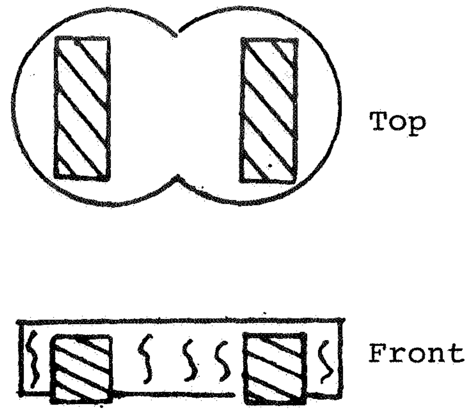


(A) SIDE TRACK

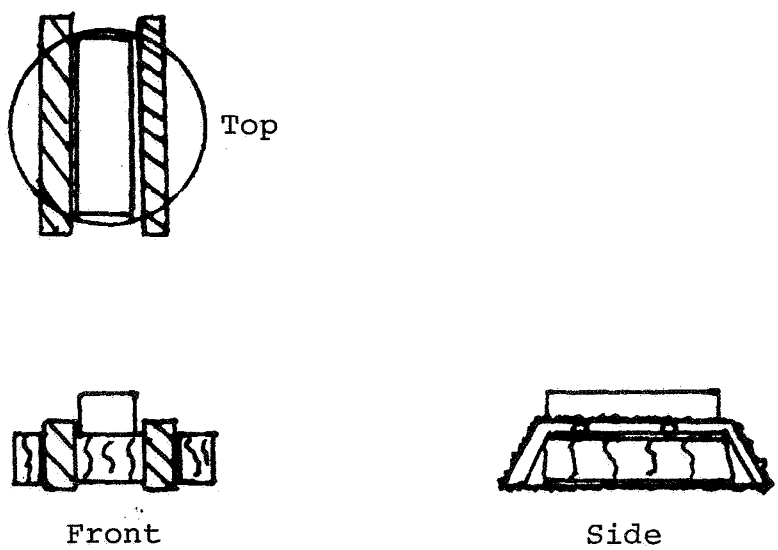


(B) CENTER TRACK

FIGURE 11.2-1 PLENUM AND TRACK LOCOMOTION CONFIGURATIONS

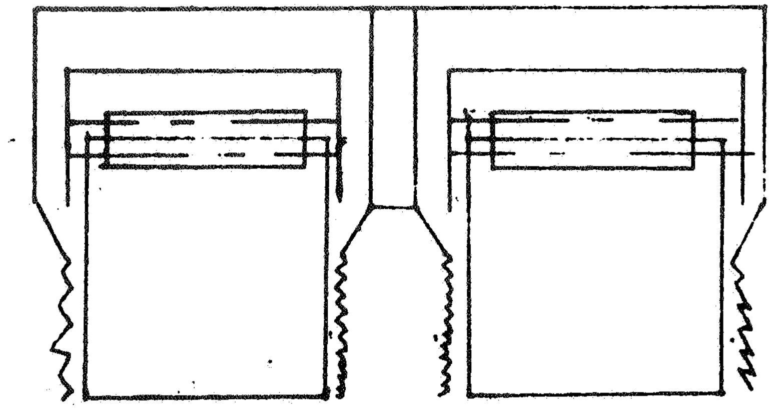


(A)



(B)

FIGURE 11.2-2. (A) FIGURE EIGHT AND (B) TRACKED WRAPPED PLENUM CONFIGURATIONS OF PLENUM AND TRACK



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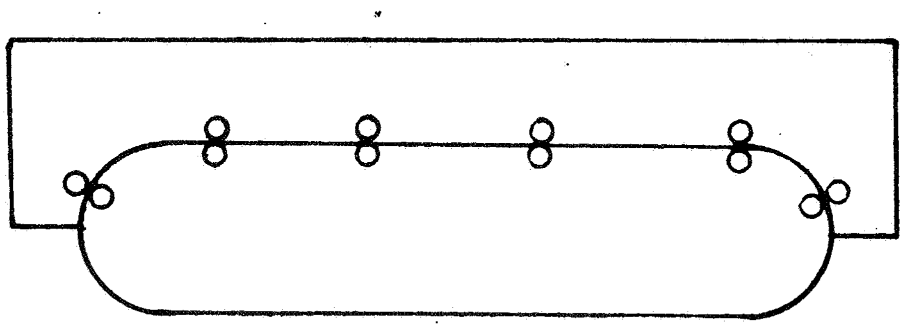


Side

FIGURE 11.2-3. GAS INFLATED TRACK CONFIGURATION

OF PLENUM AND TRACK

These five configurations were evaluated by the entire design team on the basis of certain plenum and track characteristics. The criteria for the evaluation and the results are presented in Table 11.2-1.

	SIDE TRACK	CENTER TRACK	FIGURE EIGHT	TRACK WRAPPED PLENUM	GAS INFLATED TRACK
OBSTACLE AVOIDANCE	3	0	3	5	5
DUST GENERATION	0	0	0	0	5
STABILITY	5	0	3	3	3
STRUCTURE	0	0	3	3	5
RANGE	5	5	5	5	3
STEERING & STOPPING	5	0	3	3	5
STATE OF ART	3	3	3	3	0
TOTAL	21	8	20	22	26

TABLE 11.2-1. EVALUATION OF FIVE PLENUM AND TRACK CONFIGURATIONS

Each criteria was considered by the whole design team and an effectiveness rating from 0 to 5 assigned with 5 denoting the best. The result of this team effort was that the gas inflated track be the plenum and track type configuration considered in a preliminary design.

11.2.2. Track

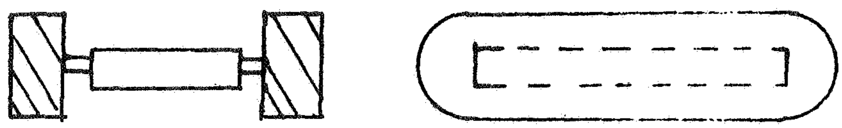
The second locomotion subsystem candidate considered was the track. The configurations incorporating only the track concept which were considered are illustrated in Figures 11.2-4 and 11.2-5.

The first decision made on the purely tracked configurations was that four tracks was nothing more than a wheeled configuration. In addition, turning a four tracked vehicle would impose difficulties so it was eliminated. The second decision was that a center platform configuration would provide a highly desirable low center of gravity. Therefore, with four tracks and top platform configurations eliminated, the two tracked center platform configuration, Figure 11.2-4 (A) would be considered in a preliminary design.

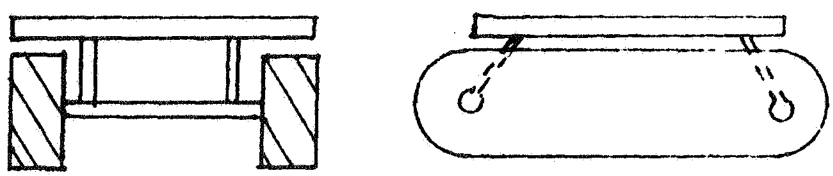
11.2.3. Track and Wheel

The third locomotion subsystem candidate considered was the track and wheel. The configurations incorporating both the track and wheel concept which were considered are illustrated in Figure 11.2-6.

In considering the one track-two wheels configuration it was noted that steering with one track would be very difficult. Therefore, the configuration which incorporates two wheels and two tracks, Figure 11.2-6, (B) would be considered in a preliminary design. The common name for a vehicle with such a locomotion subsystem is "half track".

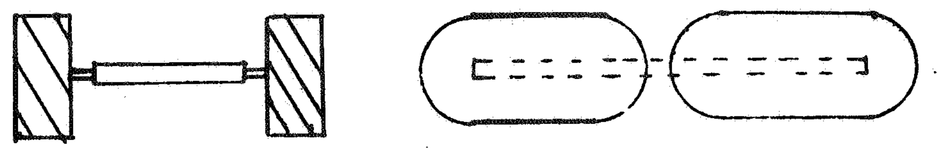


(A) Center Platform

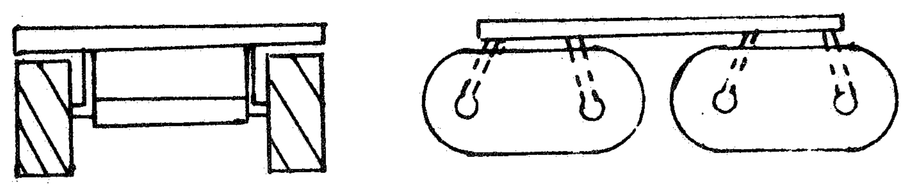


(B) Top Platform

FIGURE 11.2-4. CONFIGURATIONS INCORPORATING TWO TRACKS

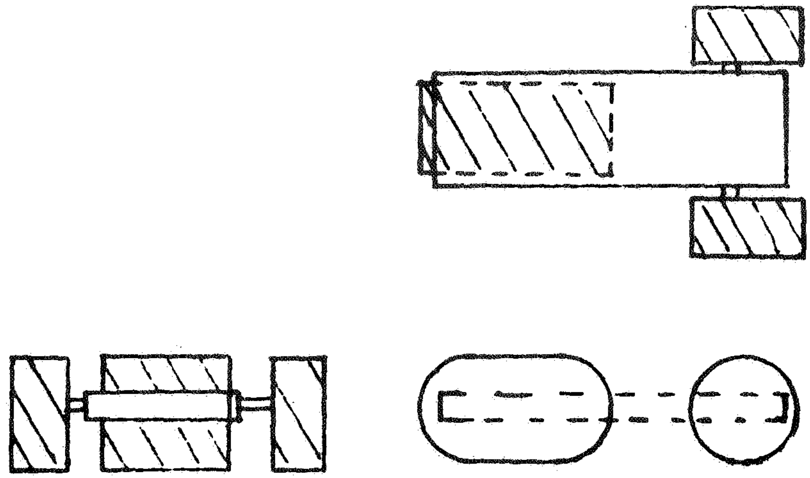


(A) Center Platform

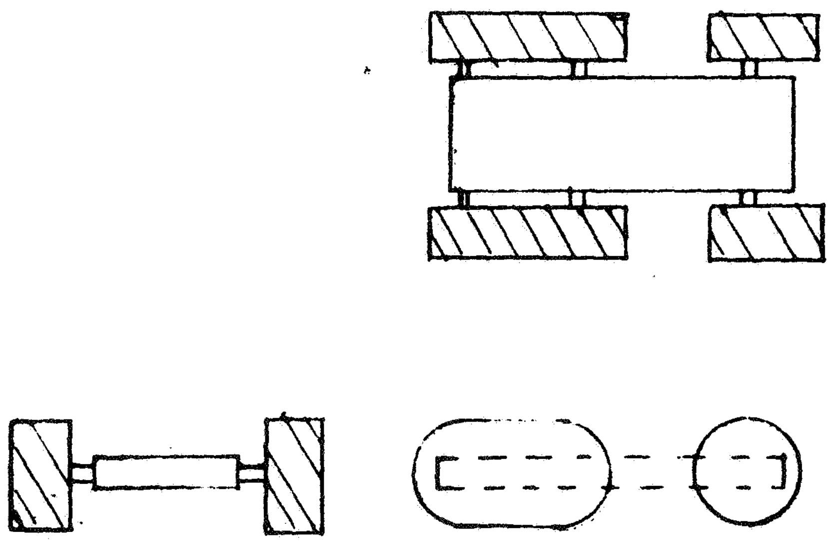


(B) Top Platform

FIGURE 11.2-5. CONFIGURATIONS INCORPORATING FOUR TRACKS



(A) One Track and Two Wheels



(B) Two Tracks and Two Wheels

FIGURE 11.2-6. CONFIGURATIONS INCORPORATING THE TRACK AND WHEEL

11.3. Preliminary Design and Evaluation of Three Candidates

11.3.1. Gas Inflated Track

11.3.1.1. Illustration of Concept

The gas inflated track incorporates two tracks which are supported by a differential pressure generated by a plenum. Figures 11.3-1, 11.3-2, and 11.3-3 illustrate a MULE utilizing a gas inflated track for locomotion. The advantage of this concept is that the load is equally distributed over the entire track-lunar surface interface. The other subsystems, astrionics, cabin and power are incorporated in the figures but only for the sake of illustration.

11.3.1.2. Evaluation

Three methods of utilizing a differential pressure to support the tracks were considered. This represents an effort to determine the feasibility of the concept and the means of implementation.

11.3.1.2.1. Method One

Method one involved the configuration previously illustrated in Figures 11.3-1, 2, and 3. Essentially it incorporates a two sectioned track in which a portion of the track moves while another portion is fixed to the structure. Figures 11.3-4 and 11.3-5 illustrate the details of this configuration.

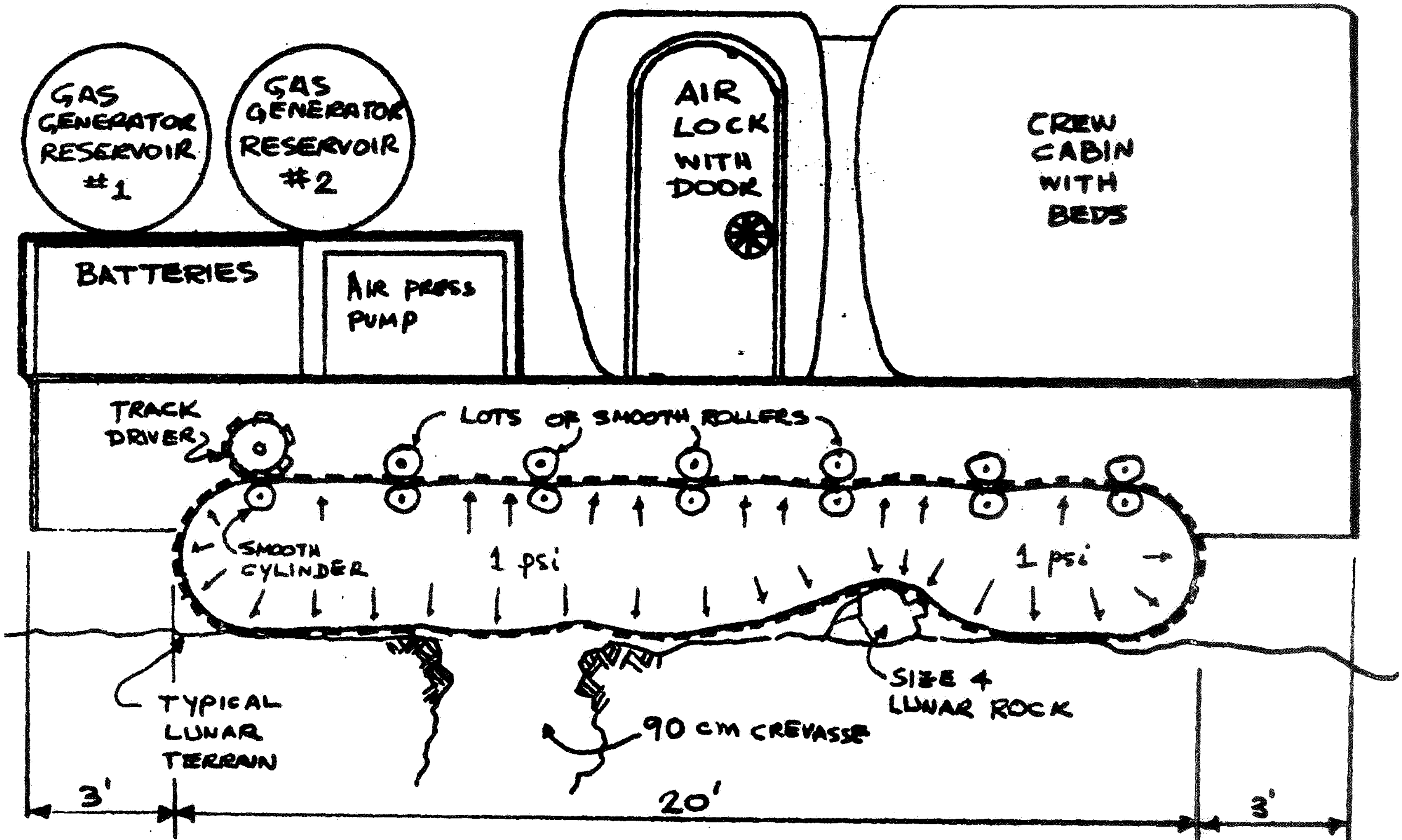


FIGURE 11.3-1. MULE WITH GAS INFLATED TRACK - SIDE VIEW

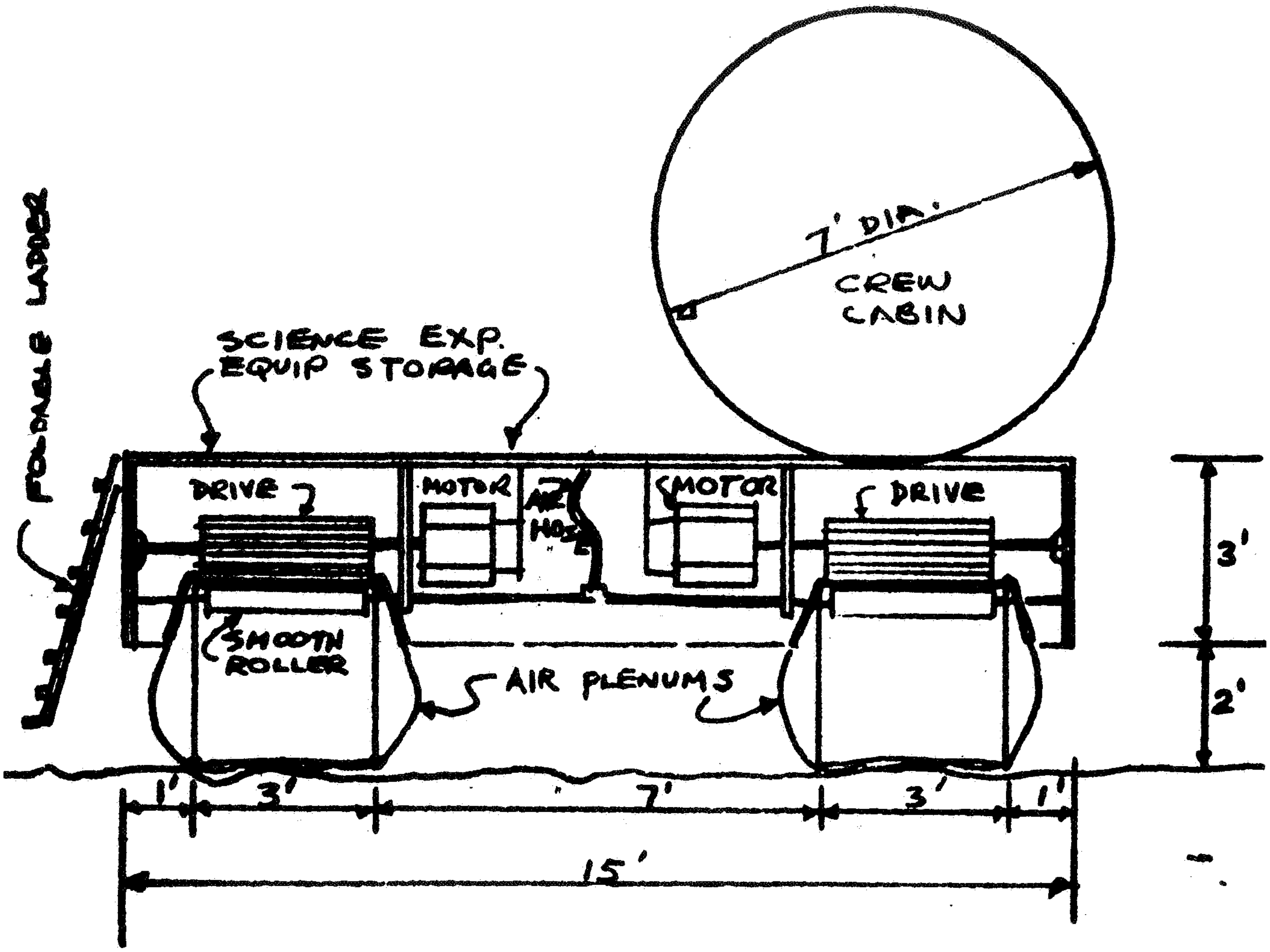


FIGURE 11.3-2. MULE WITH A GAS INFLATED TRUCK

FRONT VIEW

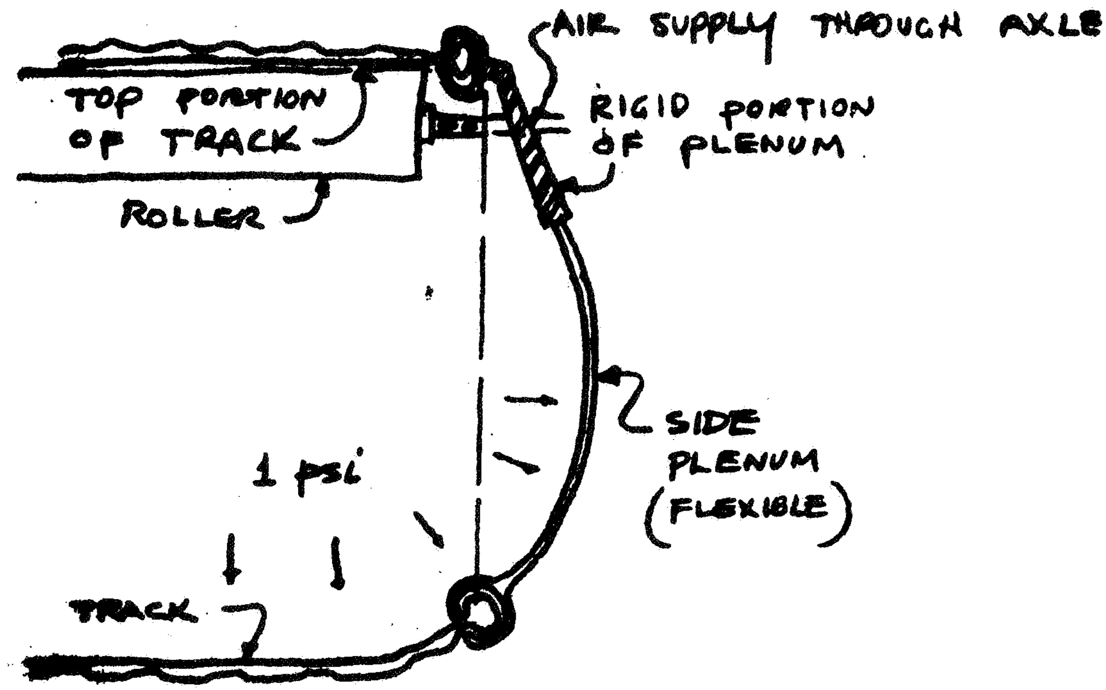
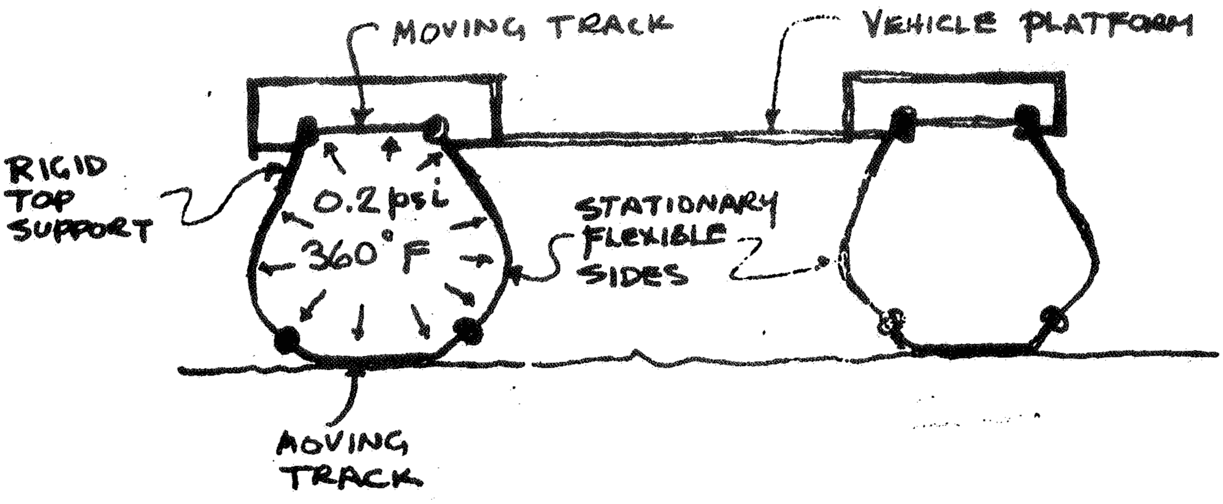
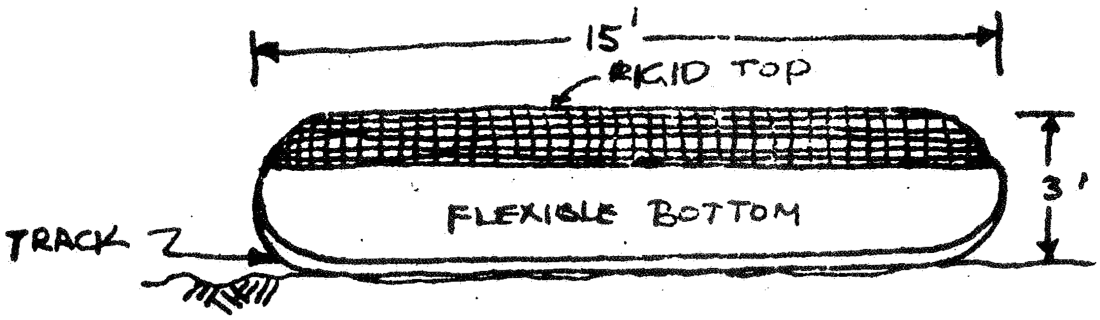


FIGURE 11.3-3. GAS INFLATED TRACK - DETAIL



Front View



Side View

FIGURE 11.3-4. TWO SECTION GAS INFLATED TRACK

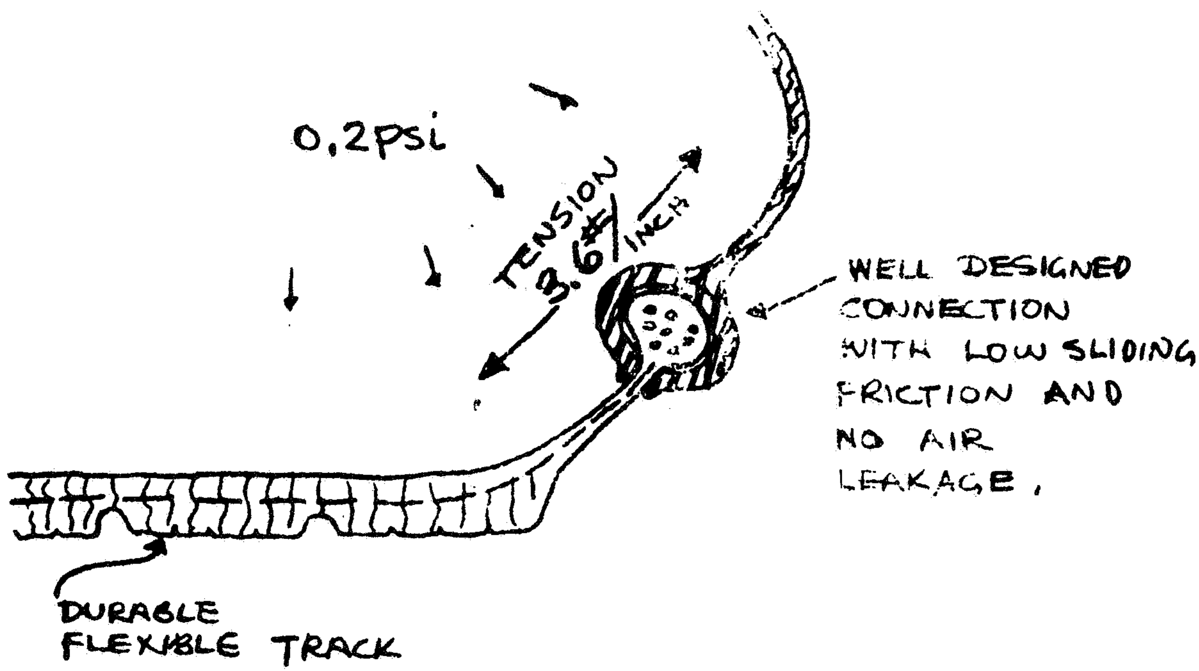


FIGURE 11.3-5. DETAIL OF TWO SECTION

GAS INFLATED TRACK

The question which must be answered about this configuration is how much power is required to overcome the friction between the moving portion of the track and supporting structure.

A. Pressure required in track

Mobility vehicle weight $= \frac{10{,}000 \text{ lbs.}}{6}$

$= 1667$ lbs. on moon

Weight on each track $= \frac{1667}{2} = 835$ lbs.

Bearing surface of 1 track $2' \times 15' = 30 \text{ ft}^2$

or $30(144) = 4320 \text{ in}^2$

Pressure required in track $= \frac{835}{4320} = 0.2$ psi

B. Tension in joint $= \frac{pd}{2} = \frac{(0.2)(36")}{2} = \frac{3.6 \text{ lbs.}}{\text{in.}}$

or 43.2 lbs/ft

C. Length of joint $15' \times 2 + 3' = 33'$

or $33' \times 2 = 66'$ for both tracks

D. Total force in joint tension

$66' \times 43.2$ lbs/ft $= 2851.2$ lbs.

E. Coefficient of friction in joint assumed to be 0.10.

F. Total drag force in each track

$2851.2 \times 0.10 = 285$ lbs.

G. Power to overcome frictional resistance

speed = 10 km/hr = 9 ft/sec

power $= FV = (285)(9) = 2565 \frac{\text{ft-lbs}}{\text{sec}}$

for two tracks, $5130 \frac{\text{ft-lbs}}{\text{sec}}$

horsepower required = $\frac{5130}{550}$ = 9.3 h.p.

11.3.1.2.2. Method Two

Method two incorporates a track on a gas inflated bag. Figure 11.3-6 illustrates this concept.

There are two possible considerations that can be made concerning this concept.

A. No air loss - sealed bag

Coefficient of friction for teflon up to 550^{o}F at 1000 fpm or 20 km/hr is between 0.16 and 0.24

Assuming f = 0.2,

Frictional force = (0.2)(1667) = 333 lbs.

horsepower required = 5.5 h.p.

B. Assume gas losses through jets with surfaces acting as an air bearing. Using a gap of 0.01" between track and gas bag the air loss is of importance.

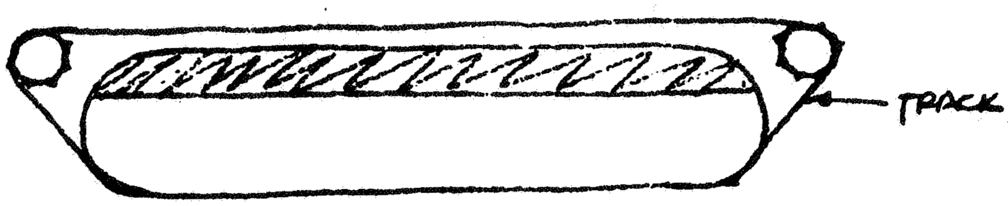
Δp = 0.2 psi

air loss = 0.36 lbs/sec

= 1296 lbs/hr

11.3.1.2.3 Method Three

Method three incorporates plenums within plenums to reduce the gas losses. Figure 11.3.1.2.3-1 illustrates this concept.



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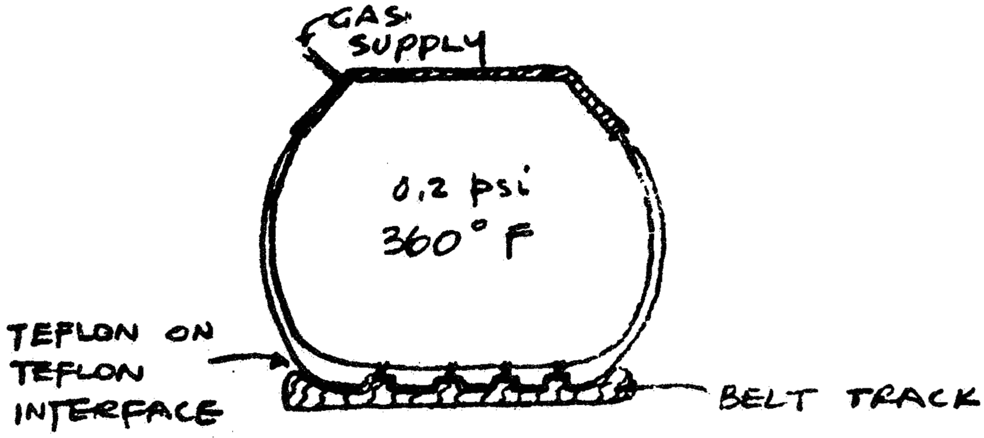

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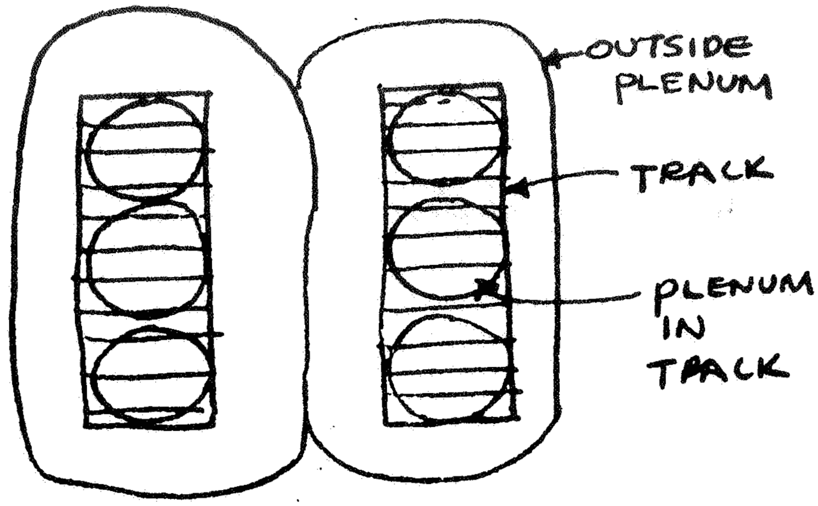
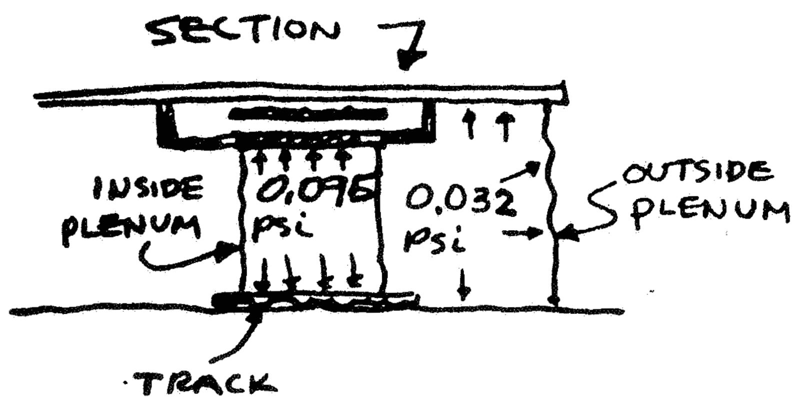
FIGURE 11.3-6. TRACK ON GAS INFLATED BAG



Top



Front

FIGURE 11.3-7. PLENUMS WITHIN PLENUM CONCEPT

The gas losses must be considered and used as a basis for evaluation.

A. Assume half load on track and half on ground from outside plenum.

B. Pressure inside plenums in track 0.095 psi

Pressure under outside plenums 0.032 psi

$\Delta p = 0.063$

C. Approximate gas losses

$$= \frac{0.063}{0.2} \times 0.36$$

$$= 0.56 \times 0.36 = 0.202 \text{ lbs/sec}$$

$$= 730 \text{ lbs/hr}$$

11.3.1.3. Results and Conclusions

The results of this analysis indicate that either large frictional forces are incurred by holding gas losses down or by reducing the frictional forces large gas losses are incurred. Some of the specific numbers follow,

Power required to overcome frictional resistance - no gas losses	9.3 hp
Gas losses incurred with air bearing - low frictional resistance	1296 lbs/hr or 730 lbs/hr

In view of the fact that to climb a 30° slope at 10 km/hr in a 1667 lb. vehicle, 13.7 hp is required, the 9.3 hp required to overcome frictional resistance of the track is significant. If on the other hand, we assume a low frictional resistance by expelling

730 lbs. of gas per hour we incur weight penalties. For example, the short 36 hour, 250 km mission previously defined requires 25 hours of driving time. At 730 lbs. per hour of gas losses, 25,250 pounds of gas would have to be generated. This obviously, is another unacceptable design.

The conclusion drawn from the analysis of the gas inflated track is that it cannot perform the required mission of the MULE. If gas losses are kept low, frictioned resistance puts an unacceptable requirement on the power subsystem. If, however, frictional resistance is kept low, the gas losses far exceed the gross weight limitations imposed on the MULE. Therefore, the gas inflated track was eliminated from further consideration.

11.3.2. Half Track

11.3.2.1. Description of Concept

The half track configuration of the MULE which was considered is illustrated in Figures 11.3-8 through 11.3-12. As the figures indicate, each wheel and track is driven separately by an electric motor. The large front wheels provide improved steering capability and are large enough to negotiate a 50 cm step. The pistons, located over the front of the tracks, allow for a decrease in track contact area when high speed and turning ease is desired.

11.3.2.2. Evaluation and Conclusion

The half track configuration provides an improvement in steering capability over the fully tracked vehicle due to a lower dragging

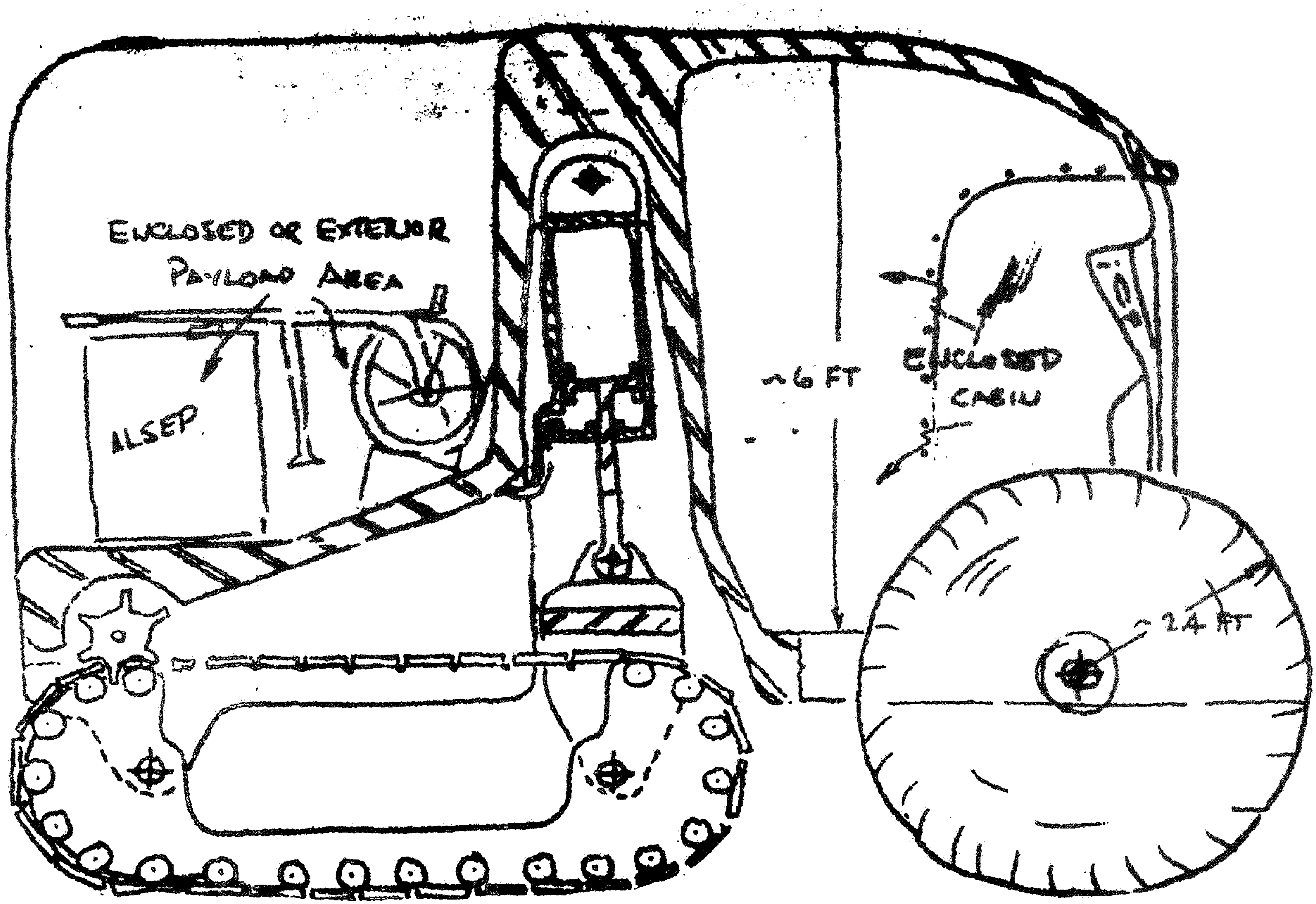


FIGURE 11.3-8. HALF TRACKED MULE - SIDE VIEW

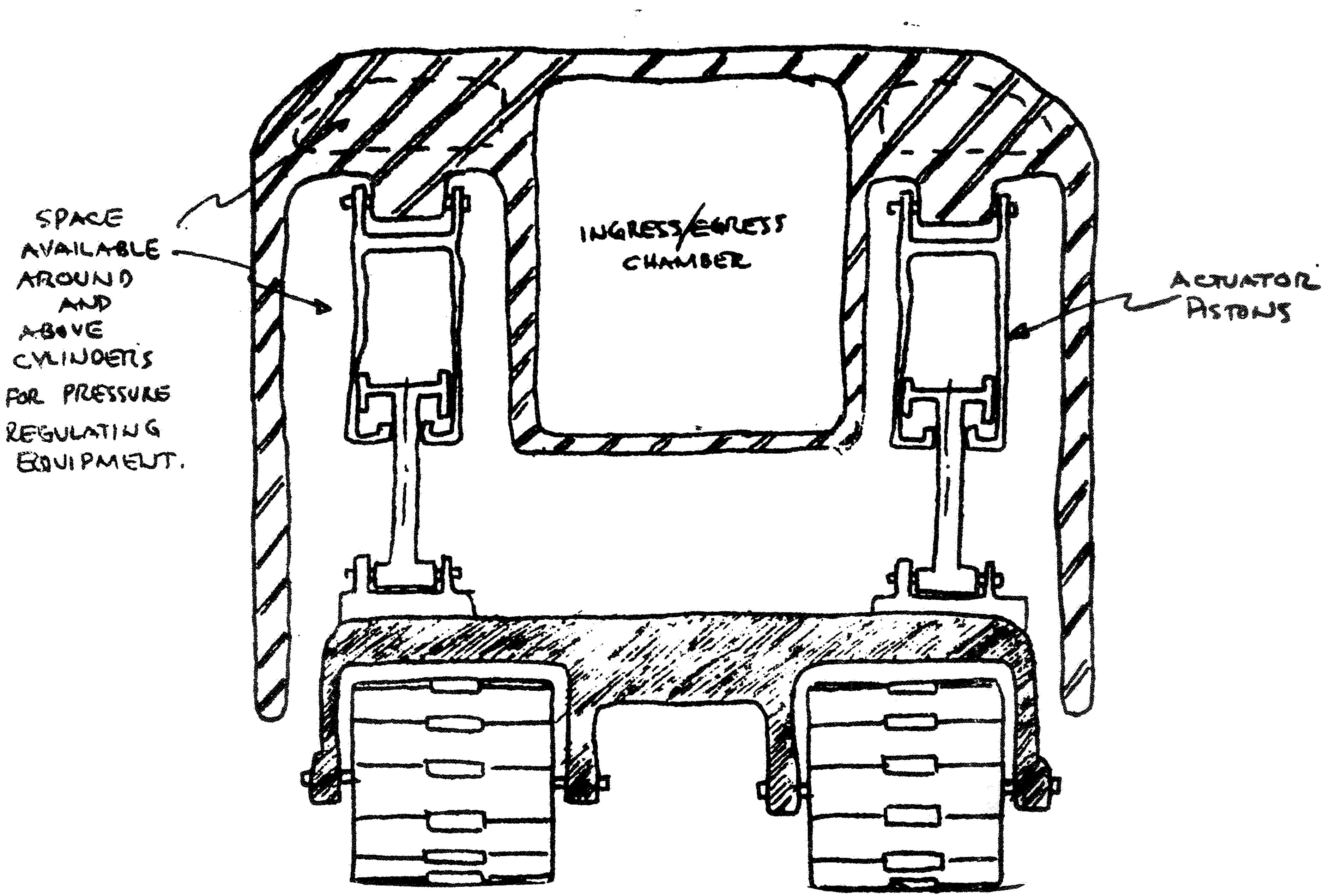


FIGURE 11.3-9. HALF TRACK MULE - FRONT VIEW

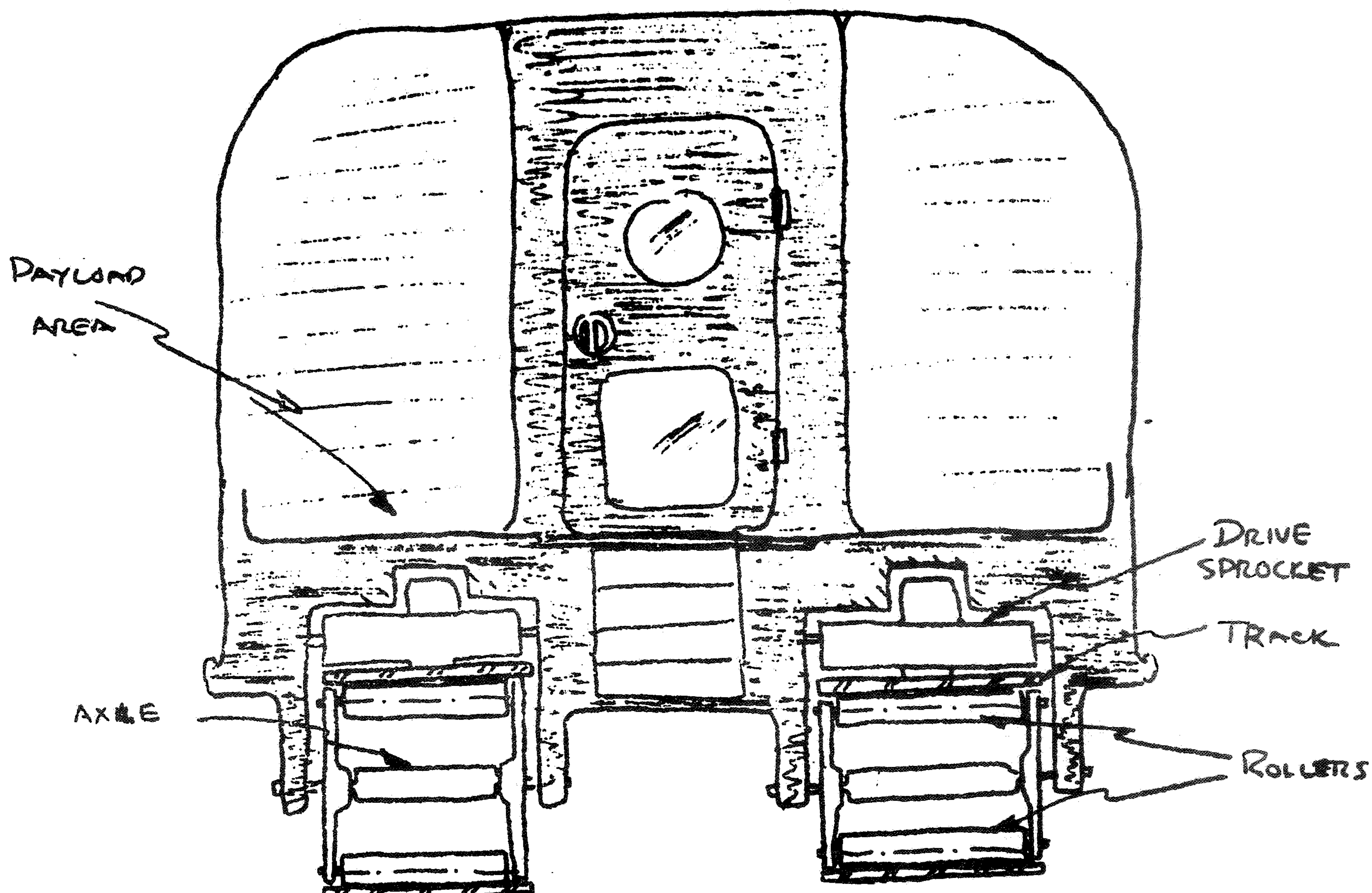


FIGURE 11.3-10. HALF TRACK MULE - REAR VIEW

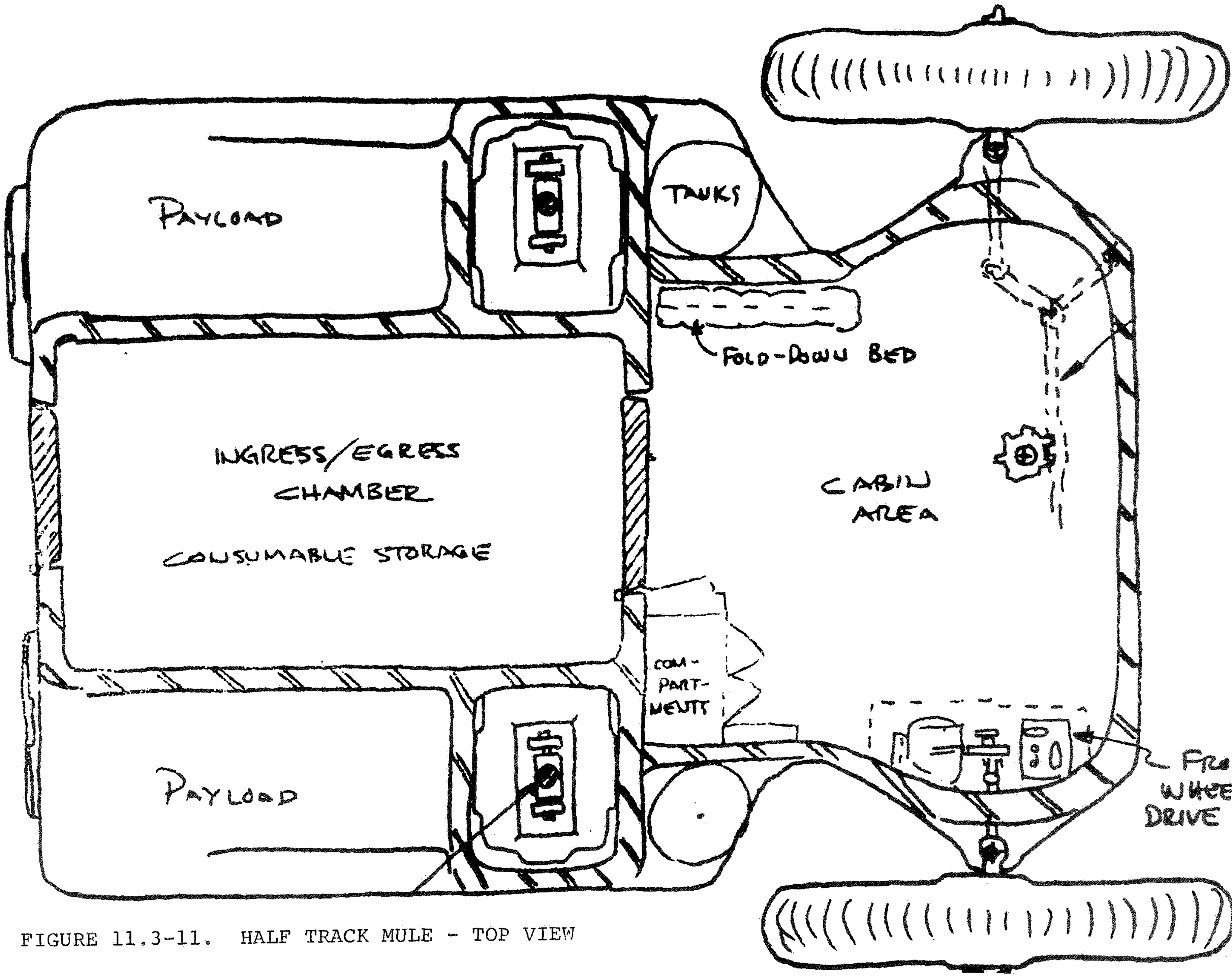


FIGURE 11.3-11. HALF TRACK MULE - TOP VIEW

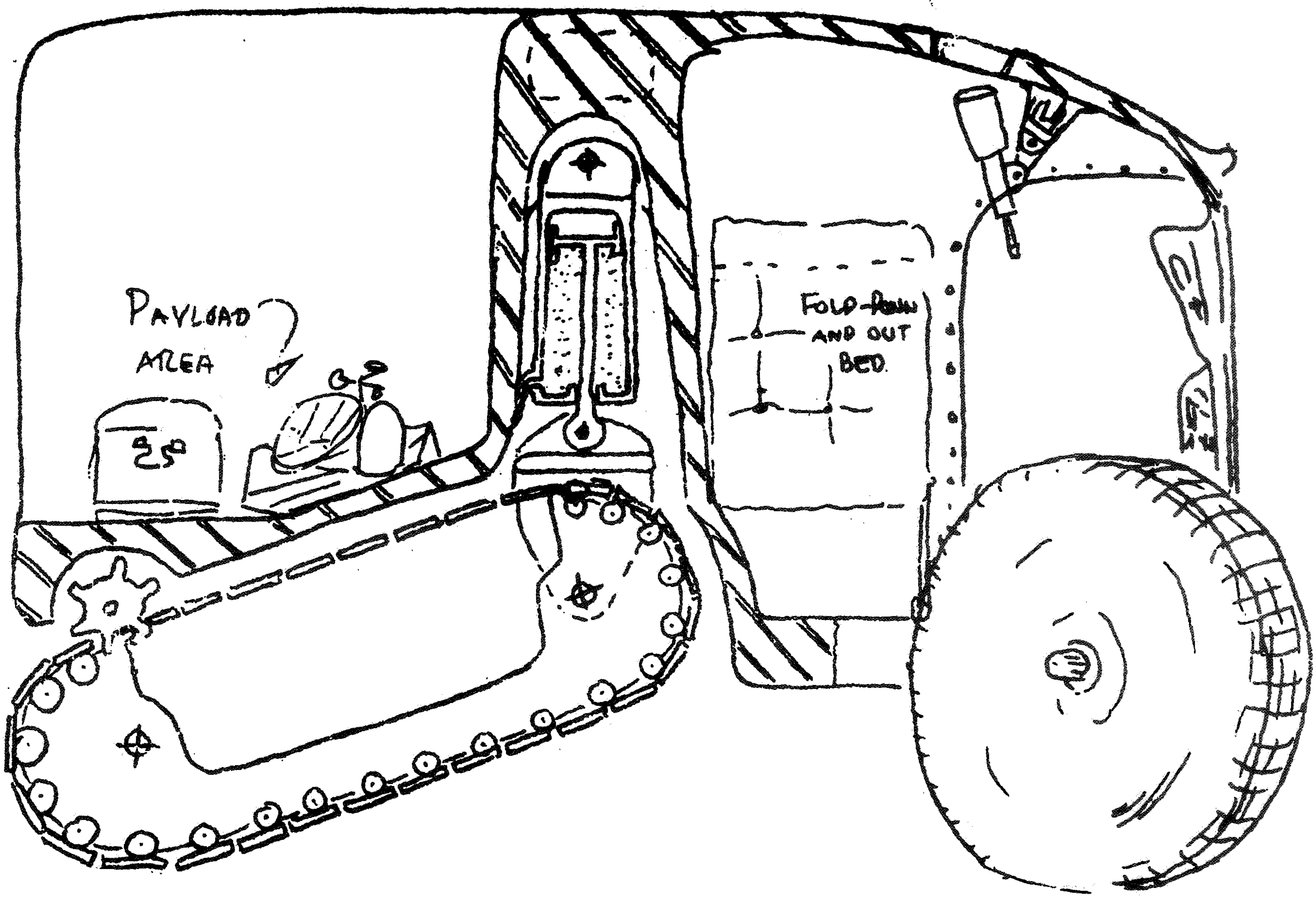


FIGURE 11.3-12. HALF TRACK MULE TURNING

resistance. It does not have the gas losses nor frictional resistance inherent in the gas inflated track. But, the expense incurred by the half track configuration lies in having two separate systems, that is, tracks and wheels. Each operates differently and, therefore, would require separate control and structure subsystems. In addition, the loss of one wheel would cause the MULE to become unoperational.

As a final point, the size of the track (length and width) would probably be as large as that of a fully tracked configuration. Considering the additional weights and controls incurred by incorporating wheels and the safety factor should one wheel fail, the half track configuration was eliminated.

11.3.3. Track

11.3.3.1. Description of Concept

The configuration considered for a tracked MULE is illustrated in Figure 11.3-13. As the figure indicates this configuration incorporates two tracks driven by four wheels. Each wheel is driven separately by an electric motor. A series of idler wheels between the main wheels distribute the load over the entire track. An idler wheel in the front of each track maintains tension and allows for the negotiation of a 50 cm step. The platform is attached between the tracks at axle level providing for a low center of gravity. The structure extends over the top of the tracks for weight distribution and dust blockage.

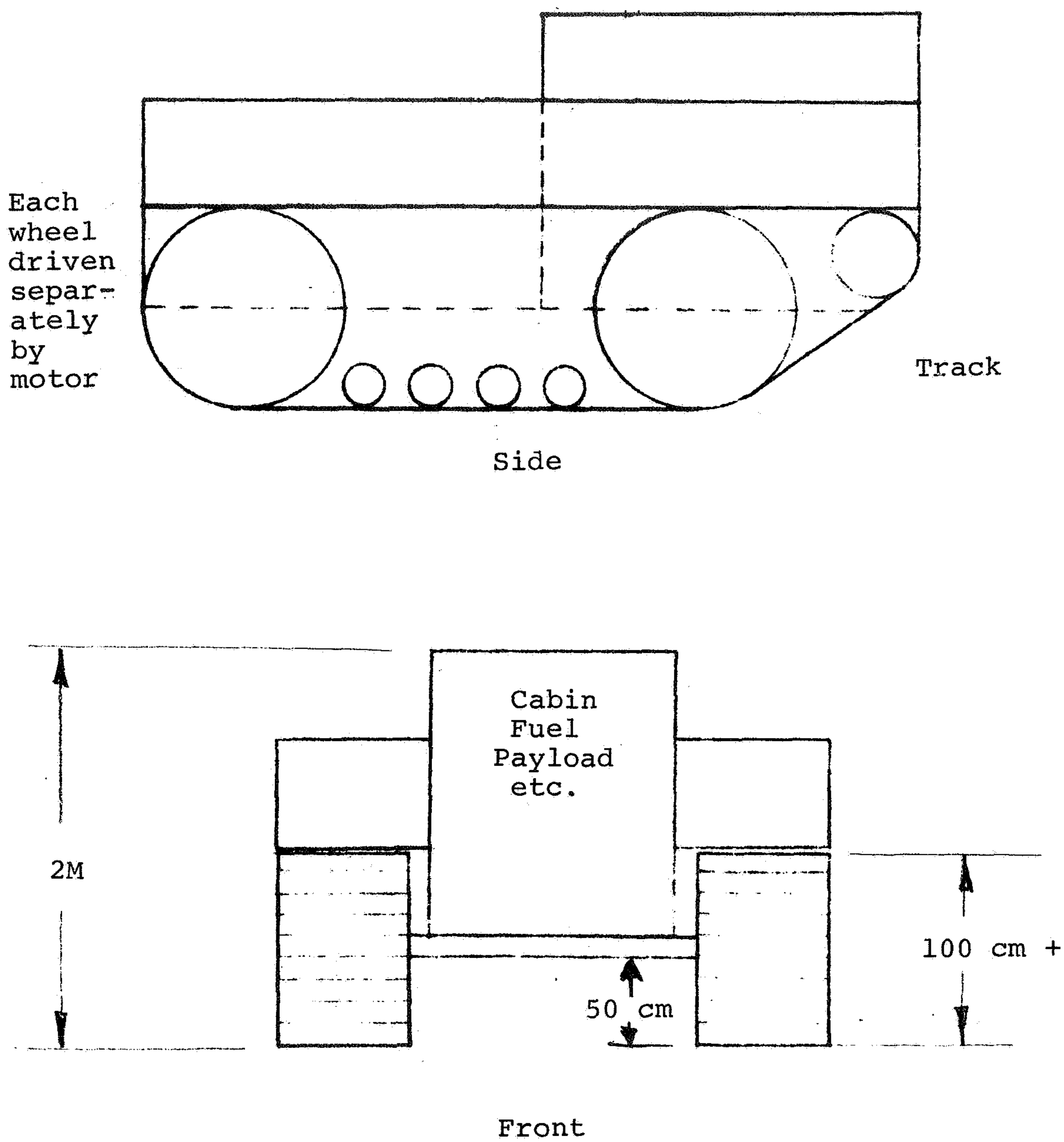


FIGURE 11.3-13. TRACKED CONFIGURATION

11.3.3.2. Evaluation and Conclusion

The purely tracked configuration has the advantages of (1) low center of gravity, (2) good distribution of load, (3) single system concept, (4) redundancy (if track breaks wheels still operational) and (5) zero turning radius. With the plenum plus track and track plus wheel configurations eliminated the recommended configuration for the final design of the MULE is a tracked vehicle.

11.4 Final System Configuration

11.4.1 Design Requirements

11.4.1.1. Statement of Work

The statement of work imposed the following design requirements on the MULE:

(1) Range of MULE 1500 kilometers unmanned

(2) Range of MULE 250 kilometers manned

(3) Payload of MULE 2000 pound plus 800 pounds for crew

(4) Lifetime of MULE one year

(5) Dry weight of MULE less than 5000 pounds

(6) MULE capable of climbing and descending 30° slope.

(7) MULE must have ground clearance of 50 centimeters

(8) MULE must negotiate 50 centimeter step.

(9) MULE must negotiate 90 centimeter crevasse

(10) MULE must be stable in pitch and role on 45° sideslope

11.4.1.2. Gross System Requirements

11.4.1.2.1. Space Shuttle

The MULE will be delivered from earth to the LOSS by the space shuttle. The payload compartment of the space shuttle is a cylinder of diameter 14.5 feet and length 30 feet. The MULE must fit this geometry.

11.4.1.2.2. Space Tug

The MULE will be delivered to the lunar surface by the Space Tug. This imposes weight and length restrictions on the MULE. The maximum payload the Space Tug can land on the lunar surface while in the manned mode is 10,000 pounds. Also, the width of the space Tug is 22 feet. Therefore, the MULE, fully loaded with fuel and payload must be less than 10,000 pounds. The largest dimension of the MULE must be less than 22 feet.

11.4.1.3. Design Group Guidelines

11.4.1.3.1. Human Factors

The MULE will have a hard cabin with an air lock. It will be capable of supporting two men for 36 hours.

11.4.1.3.2. Astrionics

The MULE will have television cameras at such positions that remote control from the LOSS, the EOSS or the MSC can be accomplished. The MULE will have the necessary antennae for the required communication links. The MULE will have manipulators for gathering lunar samples and deploying science packages.

11.4.1.3.3. Power

The MULE will utilize fuel cells, batteries and RTG's as the source of power. For this reason, tanks for LO_2 and LH_2 and radiators for the fuel cells will be provided.

11.4.1.4. Preliminary Design Evaluation

The MULE will be a two tracked vehicle with tracks driven by wheels. The support platform will be between the tracks at axle level and 50 centimeters above the surface. The wheels will be individually driven.

11.4.2. Dimensioned Drawings of the MULE

The final configuration of the MULE is illustrated in Figures 11.4-1 and 11.4-2. As these figures indicate the MULE is a two tracked vehicle with eight individually driven wheels. An idler wheel in the front of each set of wheels keeps tension in the track and allows for the negotiation of a 50 centimeter step. The MULE has a hard cabin up front with a hard air lock behind.

11.4.3. Subsystem Locations

11.4.3.1. Locomotion

The locomotion subsystem consists of the following components:

(1) 8 - 4 foot diameter wheels

(2) 8 - 1 horsepower electric motors

(3) 8 - torsion bar suspension units

(4) 2 - 47 feet by 2 feet tracks

(5) 2 - 2 foot diameter idler wheels

The locomotion subsystem components locations are illustrated in Figure 11.4-3.

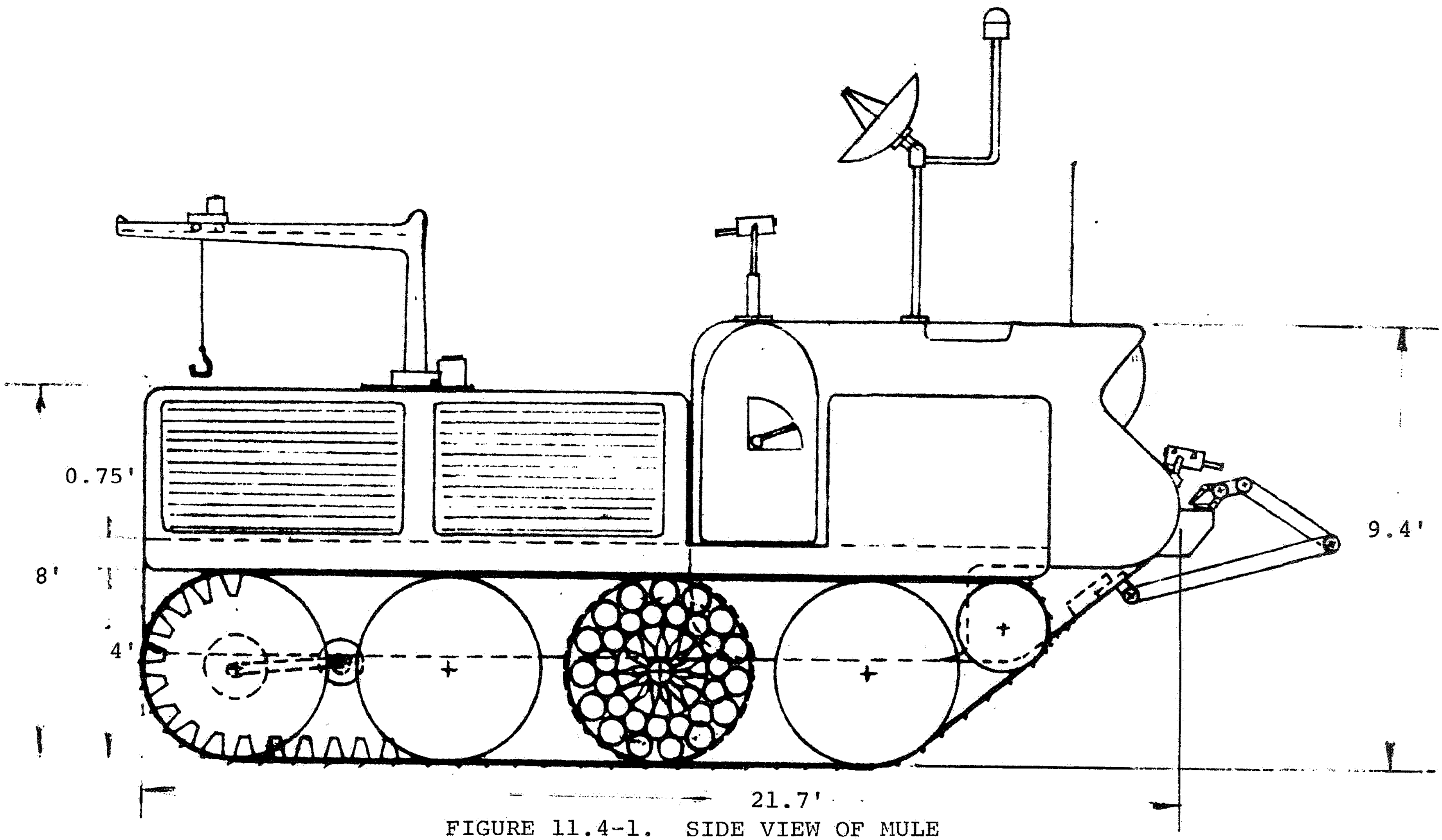


FIGURE 11.4-1. SIDE VIEW OF MULE

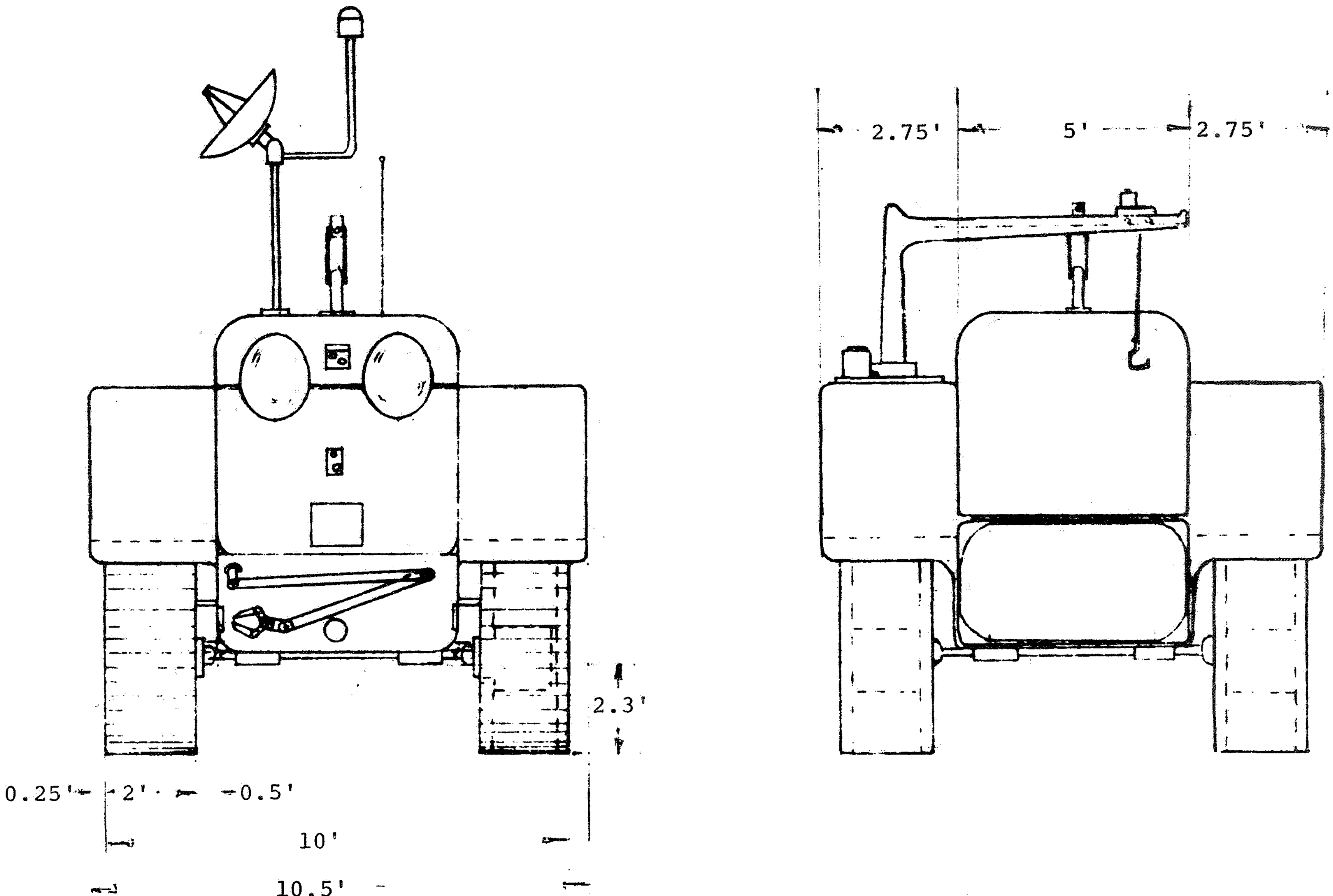


FIGURE 11.4-2. FRONT AND REAR VIEW OF MULE

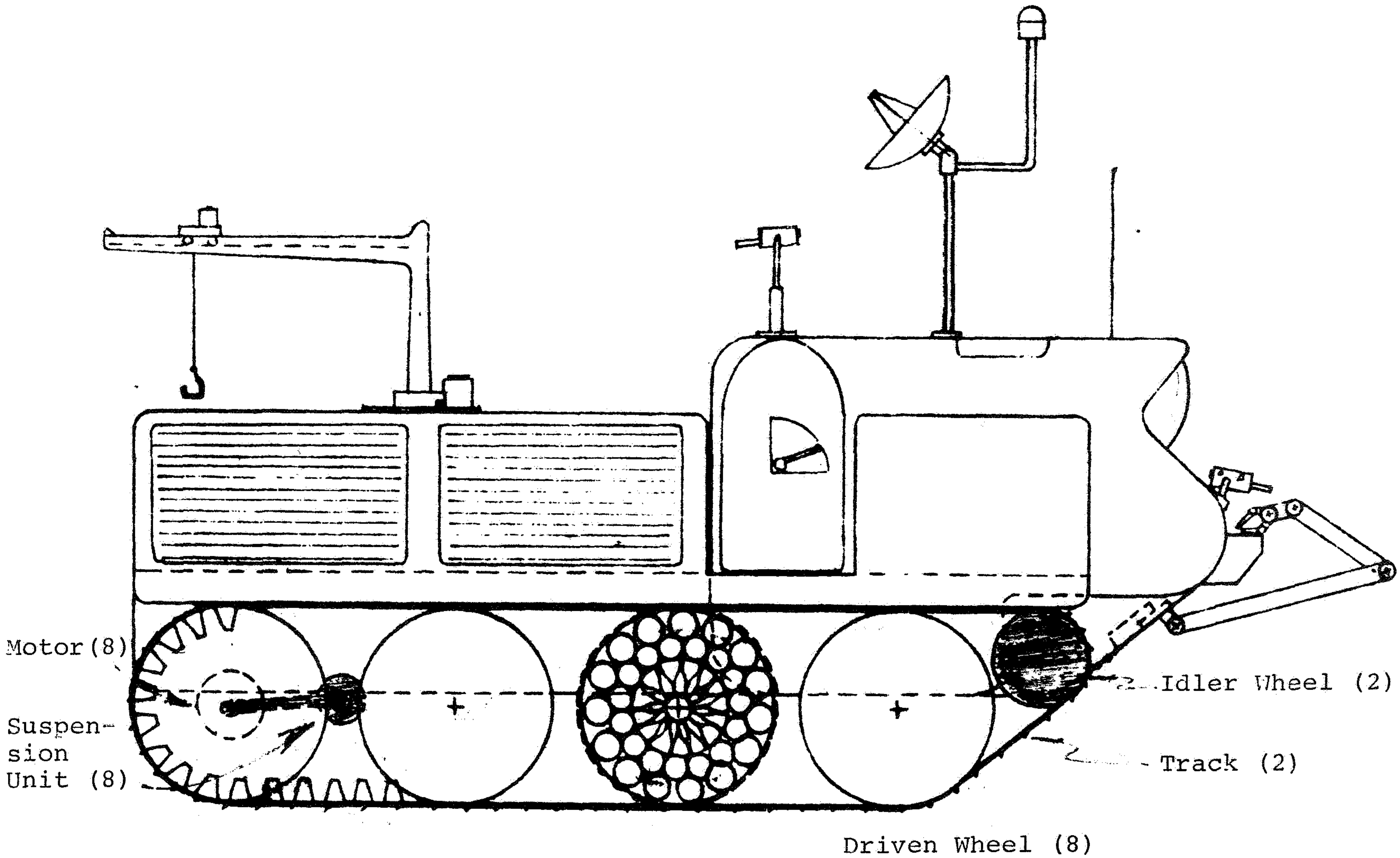


FIGURE 11.4-3. LOCOMOTION SUBSYSTEM

11.4.3.2. Power

The power subsystem consists of the following components:

(1) Fuel cells

(2) Batteries

(3) RTG's

(4) LO_2 and LH_2 tanks

(5) Fuel cell radiator panels

The power subsystem component locations are illustrated in Figures 11.4-4, 11.4-5 and 11.4-6.

11.4.3.3. Cabin

The cabin subsystem is illustrated in Figures 11.4-7, 11.4-8, and 11.4-9. The main cabin has the following dimensions; 5 feet in height, 10 feet in length and 5 feet in width. The air lock is 7 feet high, 5 feet wide and 3 feet long.

11.4.3.4. Astrionics

The astrionics subsystem consists of the following:

(1) Electronics package

(2) Omni-directional antenna

(3) Whip antenna

(4) Directional S-band antenna

(5) Forward television camera

(6) Panning television camera

(7) Manipulator

(8) Service crane

(9) Obstacle avoidance sensor

(10) Head light

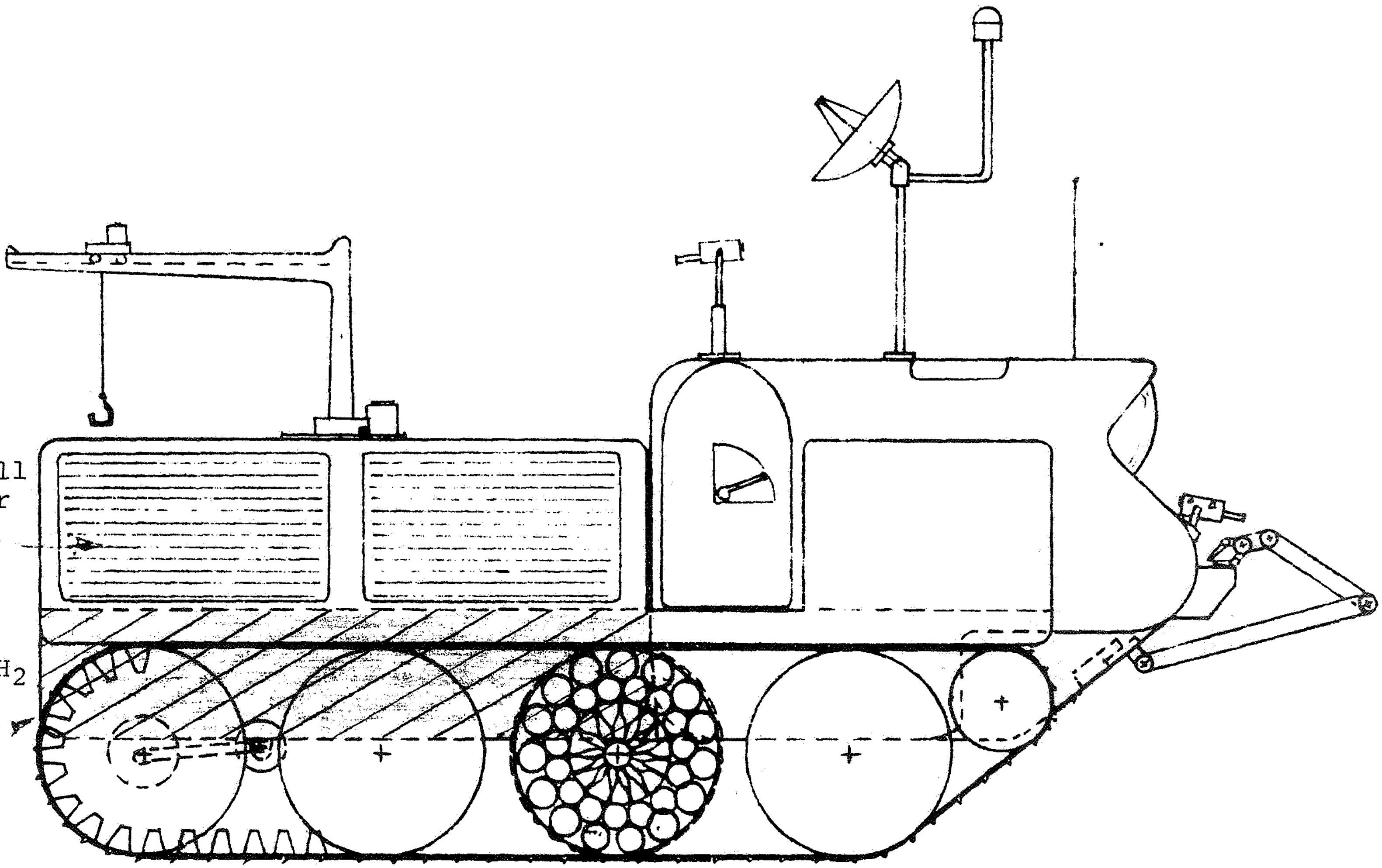


FIGURE 11.4-4. POWER SUBSYSTEM - SIDE VIEW

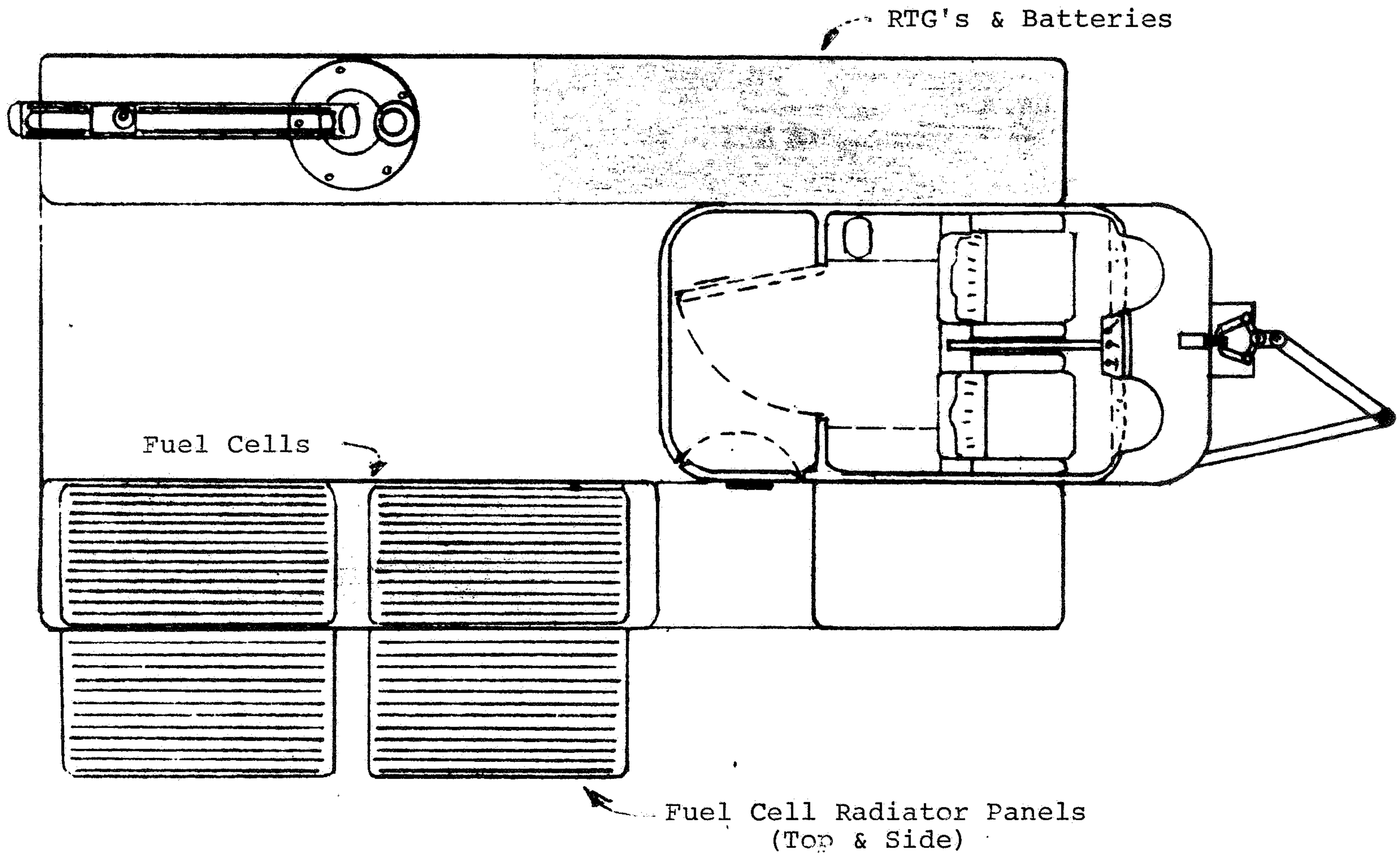


FIGURE 11.4-5. POWER SUBSYSTEM - TOP VIEW

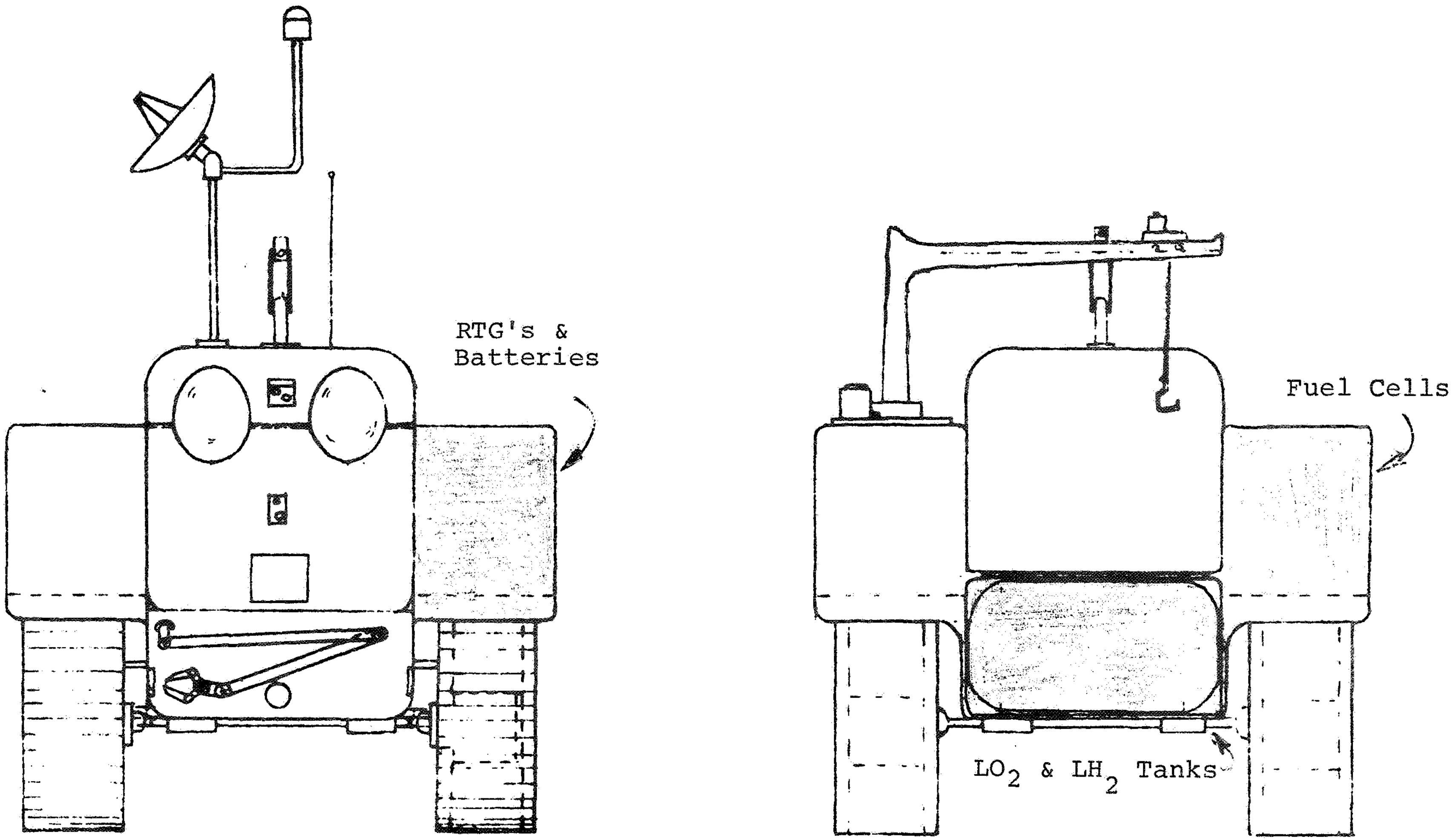


FIGURE 11.4-6. POWER SUBSYSTEM - FRONT AND BACK VIEWS

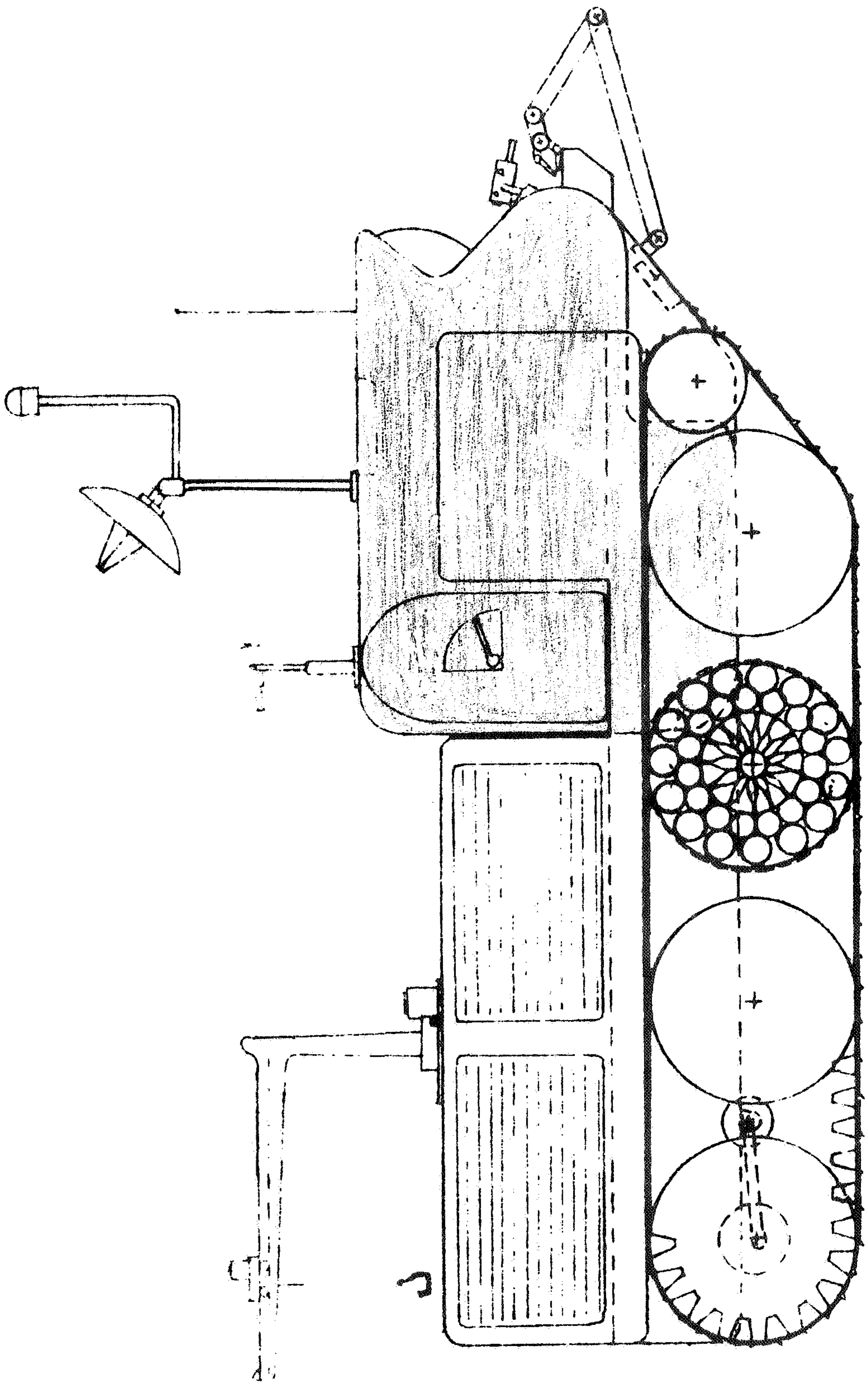


FIGURE 11.4-7. CABIN SUBSYSTEM - SIDE VIEW

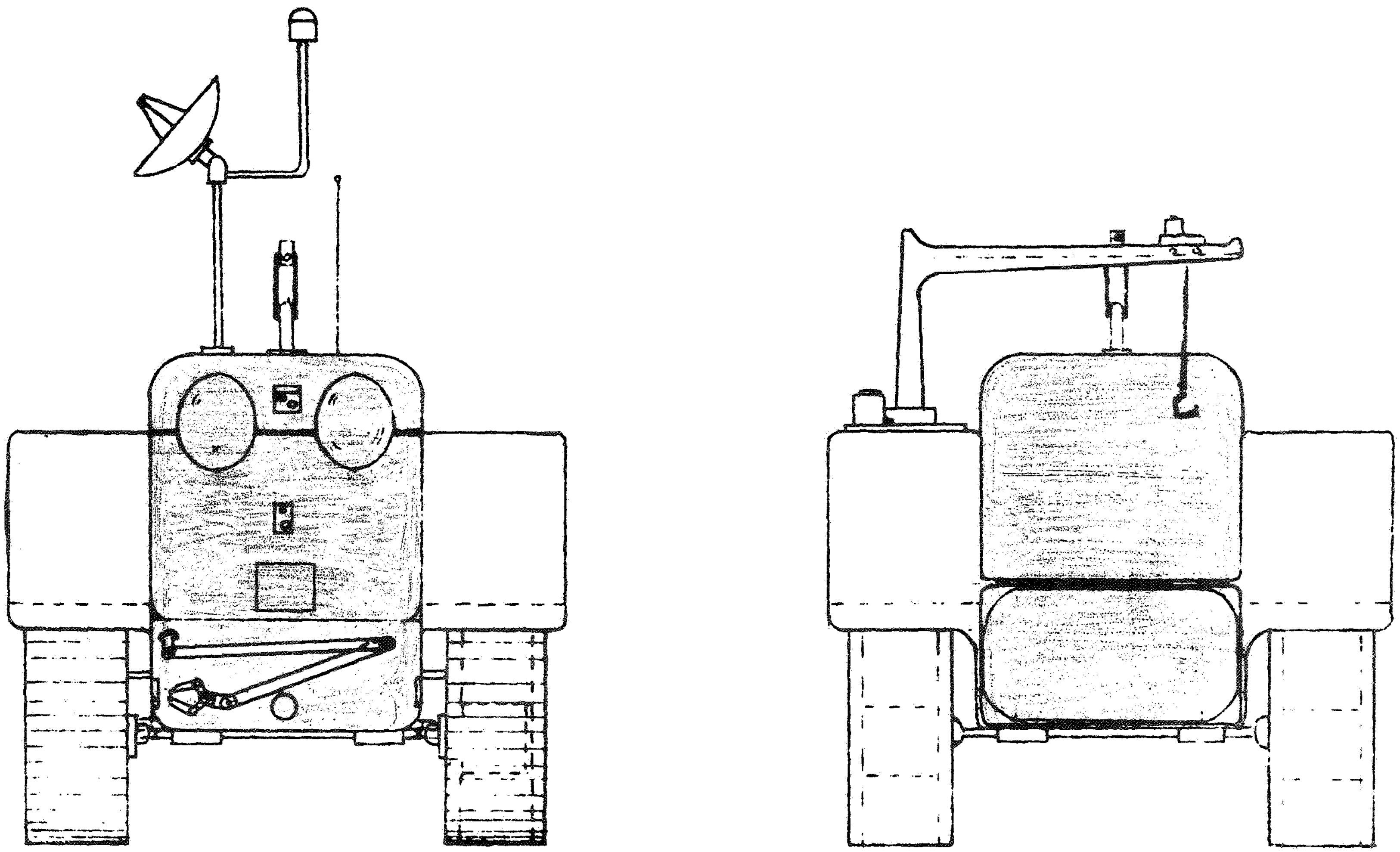


FIGURE 11.4-8. CABIN SUBSYSTEM - FRONT AND REAR VIEW

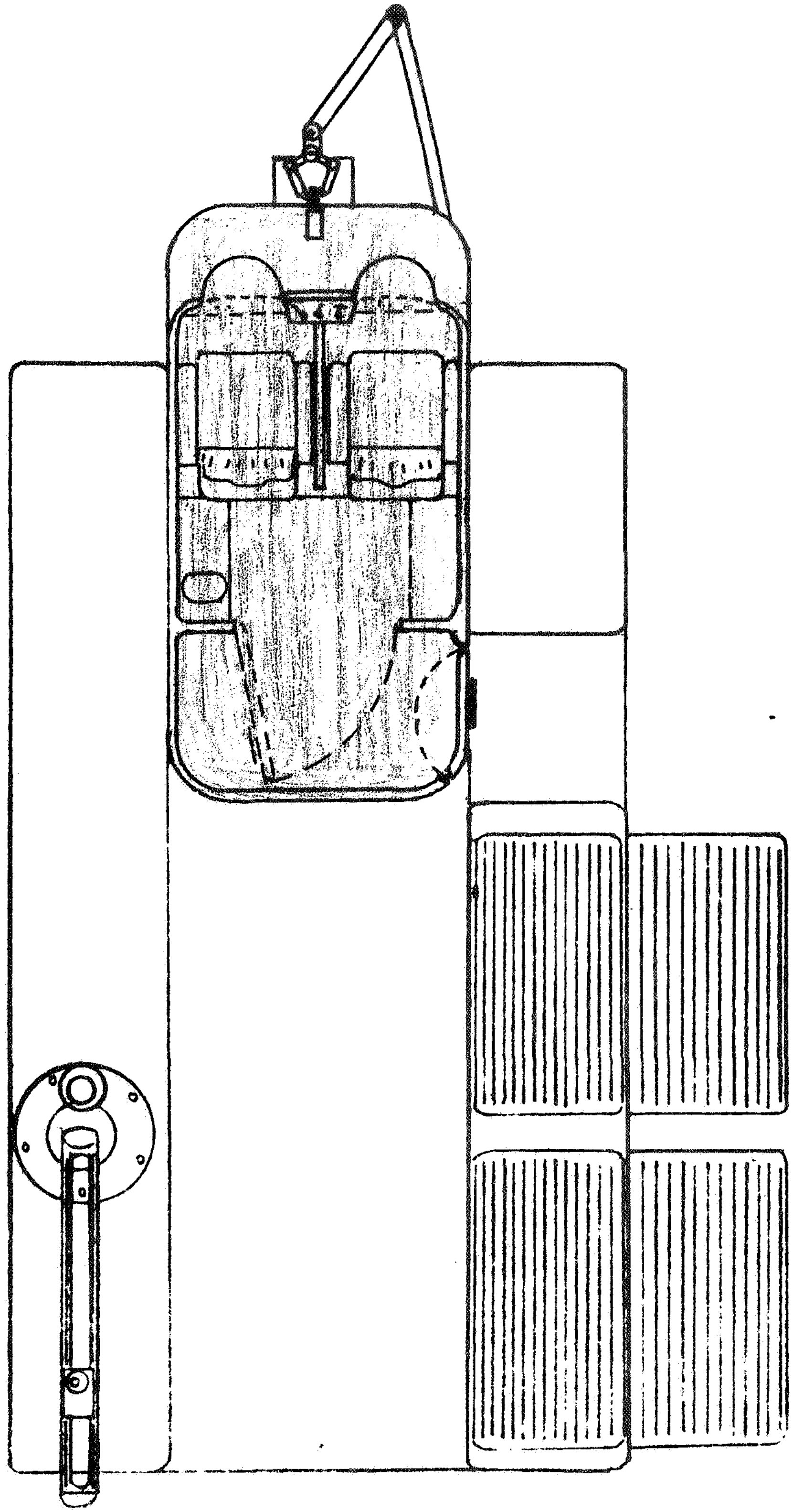


FIGURE 11.4-9. CABIN SUBSYSTEM - TOP VIEW

The locations of these components are illustrated in Figures 11.4-10 and 11.4-11.

11.4.4. Weights and Center of Gravity Specifications

11.4.4.1. Weights

The weights of subsystems and subsystem components are presented in Table 11.4-1.

ITEM	WEIGHT (LBS)
WHEELS	480
TRACKS	500
MOTORS	260
IDLERS	80
SUSPENSION	320
CABIN	1550
STRUCTURE	950
CRYOGENIC TANKS	800
BATTERIES FUEL CELLS RTG'S	460
RADIATORS	85
ASTRIONICS & MANIPULATORS	500
DRY WEIGHT	5985

TABLE 11.4-1. WEIGHT DISTRIBUTION OF MULE

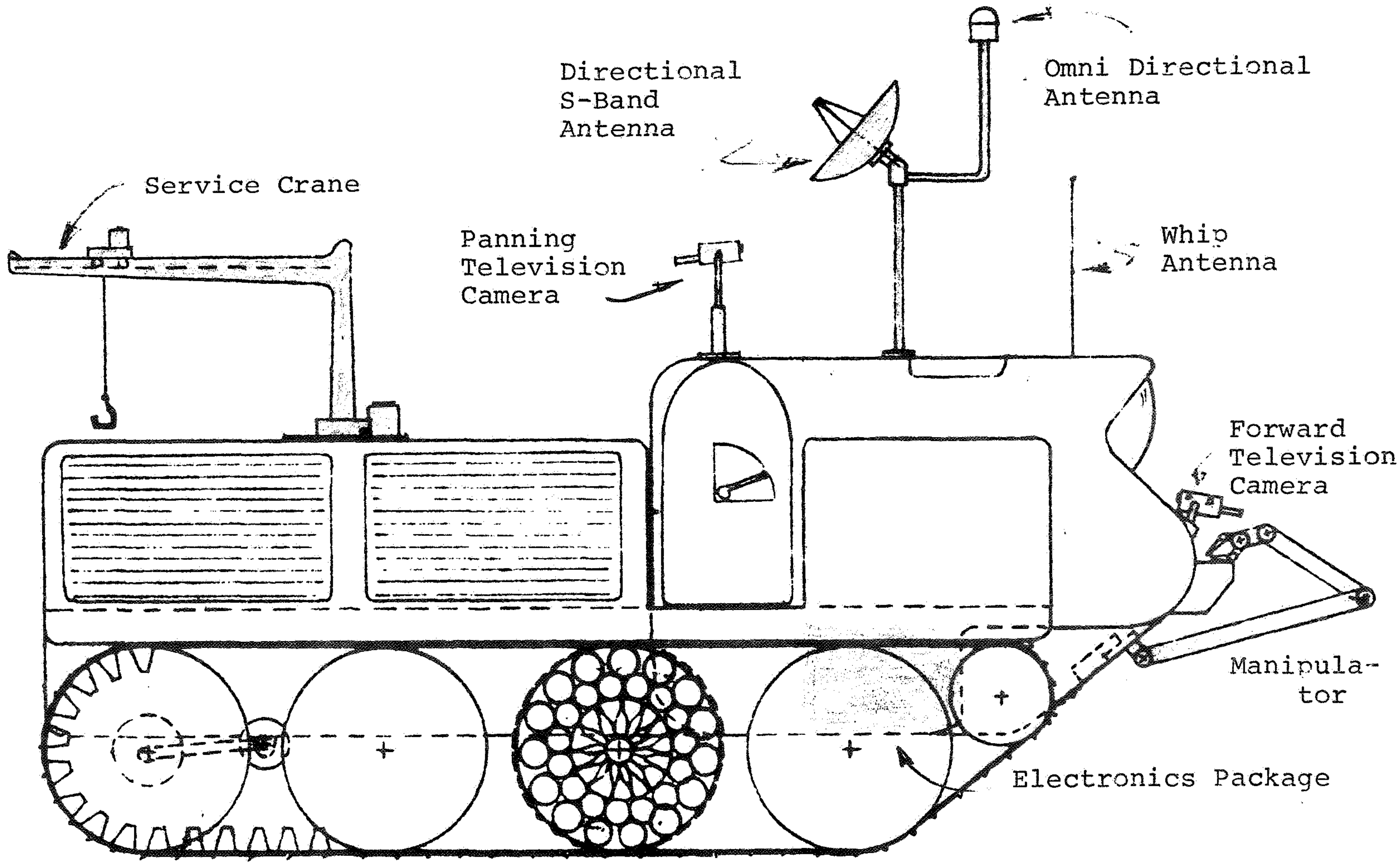


FIGURE 11.4-10. ASTRIONICS SUBSYSTEM - SIDE VIEW

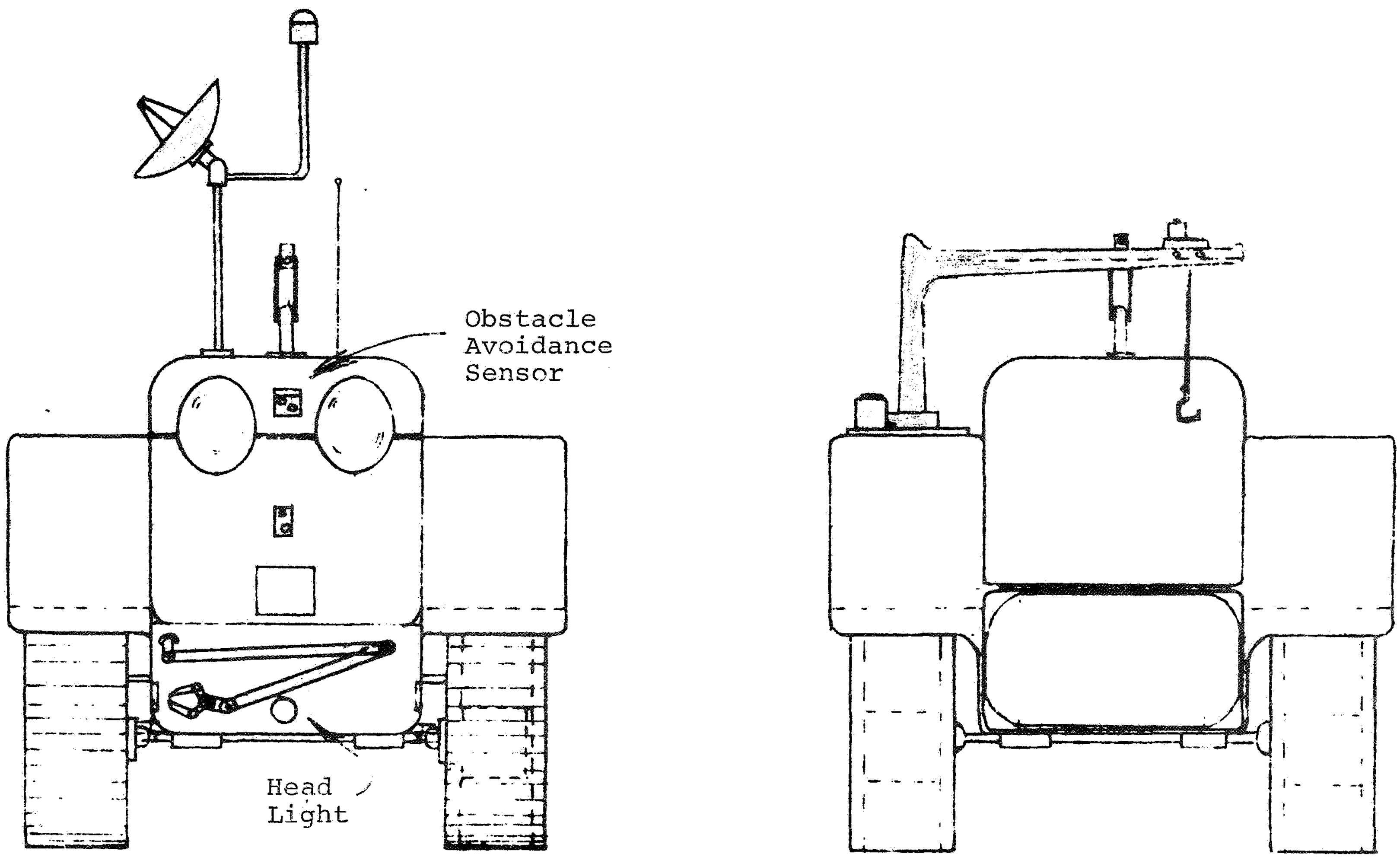


FIGURE 11.4-11. ASTRIONICS SUBSYSTEM - FRONT AND REAR VIEWS

As indicated, the dry weight of the MULE is 5,985 pounds.

11.4.4.2. Center of Gravity Specifications

The variations in the lateral center of gravity under all loading conditions are presented in Table 11.4-2.

CONDITIONS	WEIGHT (LBS)	MOMENT (FT/LBS)	C.G. (FT)
Dry MULE	5,985	26,530	4.27
MULE plus 2000 pounds of payload	7,985	38,530	4.83
MULE plus payload plus 920 pounds of fuel	8,905	42,210	4.75
MULE plus payload plus fuel plus 800 pounds for crew	9,705	47,010	4.85

TABLE 11.4-2. VARIATIONS IN LATERAL CENTER OF GRAVITY

As indicated, the lateral center of gravity varies from 4.27 to 4.85 feet above the bottom of the track. The stipulation for stability on a 45° sideslope with a track width of 10 feet requires that the lateral center of gravity be below 5 feet. Therefore, the MULE will be stable under all loading conditions on a 45° sideslope.

CHAPTER 12

LOCOMOTION SUBSYSTEM

Miltiadis Leptourgos

12.1. Introduction

The synthesis of parts to produce a structure involves the utilization of art and science in the planning, analysis, design and construction of structures. Planning consists of the development of a general layout of the structures that satisfies functional, economical and some times esthetic requirements. Analysis is the process of constructing a mathematical model for the real structure and seeking the numerical solution of the idealized problem. Design involves the selection of proper materials for the structure and the determination of the shapes, sizes and weights of the components. Construction pertains to fabrication, erection and inspection - testing of the structure. The final completion of a safe but efficient and economical structure depends on the satisfactory execution of the project through all the above stages.

It should be noted, however, that these basic steps are by no means distinct, but rather are interdependent. Some of them involve mathematical analysis; others require judgment based on experience. Quite frequently it is necessary to have a balanced combination of both, depending on the degree of complexity of the structure.

In the planning phase of the project, the major requirements are creativity, imagination and sound judgment. Experience is often the guide for selecting the type of structure. On the other hand, one should be aware of both the virtues and the weaknesses of the proposed structure.

In the analysis part, a great deal of judgment is involved in making the simplifying assumptions, which can only be verified by experience. At this stage, three distinct phases should be recognized. First, the real structure may be divided into appropriate component systems that can be analyzed separately. Second, the determination of loads that are transmitted and resisted by each system, considering not only their magnitude, location and distribution, but also the frequency of occurence and nature of their action. Third, it is necessary to examine the characteristics of the system against the consequences of failure or unserviceability of the structure. Thus, the concept of structural safety will not be lost in the idealization. These three phases are closely related to each other and all three of them affect the nature of analysis. Although the analysis itself is a mechanical procedure, the consideration of these factors involves a great deal of judgment.

In the design phase, the choice of materials for the structure depends not only on the suitability of the materials, but also on their availability; for this reason the materials should be determined in the early planning stage. The proportions of the component members of a structure must be so chosen that the structure as a whole can safely and economically perform the intended services under the prescribed loading conditions. Failure in certain parts of the structure may have local effects only, but in other cases it may cause the complete collapse of the structure. For this reason no member or connection in a structure should be overlooked. Because of the elements of uncertainty, and perhaps ignorance of the expected loads, the material properties, and the criteria for failure, a factor of safety is introduced in proportioning the structural members.

It should be observed that design is almost inseparable from analysis. The exact sizes of structural members can be determined only after a complete analysis of the structure. On the other hand, there are certain things in analysis that depend on the result of design. For example, the weight of the structure, which cannot be determined exactly until all members of the structure are proportioned.

Construction is the final stage where the work shifts from design to practical implementation; fabrication and assembling of parts takes place in this stage.

The cost of construction is a direct function of the materials used, techniques employed in obtaining the desired shape, time of fabrication, quantity of parts and other parameters. The technique of construction also has a direct effect on analysis and design. For instance, certain members or sections of a structure may be stressed to allowable stresses due to the method of assemblying the structure.

The foregoing discussion clearly indicates the importance of an integrated knowledge of structures in order to obtain the best possible solution of a structural problem. Inasmuch as engineering is an art, it cannot be completely replaced by an exact science. Engineers have learned to rely more and more on the methodology of science. But they cannot entirely dispose with judgment. Intuitive perception and insight are invaluable, but not infallible. Hence, judgment based on experience must be tempered by rational analysis. A proper balance of theoretical knowledge and practicality based on a sense of judgment, is essential for the successful design and fabrication of a structure.

12.2. Locomotion Analysis

S. J. Clark

12.2.1. Introduction

An off-the-road vehicle must be capable of; (1) developing drawbar pull for climbing slopes and obstacles, (2) maintaining a certain speed, (3) stopping in a reasonable distance, (4) providing a definite degree of stability and controllability, (5) providing a certain comfort index, and (6) carrying a certain payload. All of these functions are affected by the soil strength and the roughness and geometry of the terrain. Off-the-road vehicle design must consider the soil-vehicle interactions or the design is meaningless. The design of a vehicle for use on the lunar surface is no exception. It is in fact, more important to utilize every available bit of information regarding lunar soil and terrain parameters since we as yet have no lunar vehicle test experience.

12.2.2. Drawbar Pull

The pull or push that an off-the-road vehicle can develop for climbing slopes or pulling a load is a function of the force required to shear the soil under the vehicle footprints, the rolling resistance of the vehicle due to sinkage, and frictional energy losses.

Assuming that the thrust H and the total rolling resistance, R_t, can be separated, a simple relationship for drawbar pull is;

$$DP = H - R_t$$

12.2.3. Traction Device Thrust

The maximum thrust H that a traction device can develop is:

$$H = AC + W \tan \phi$$

where

A = area of traction device soil contact area.

C = soil cohesion, psi

W = the normal load on the traction device.

The above equation applies to the maximum thrust H developed at optimum slip. A simplified equation that includes percent slip is:

$$H = (A\ C + W \tan \phi)\ (1 - e^{-\frac{i\ell}{K}})$$

i = percent slip

K = an experimentally determined slip-strength parameter.

ℓ = foot print length

12.2.4. Rolling or Motion Resistance

Motion resistance to vehicle movement over soil is due to energy losses from soil compaction, pushing soil ahead of the wheels or tracks (bulldozing), resistance due to slope ($W \sin \beta$), and mechanical friction losses in the soil contact supporting mechanism. The equation for the resistance due to compaction is:

$$R_C = \frac{\ell}{(n + 1) \ (K_C + b \ K_\phi)} \left(\frac{W}{\ell} \right)^{\frac{n+1}{n}} \qquad (12.2\text{-}1)$$

where

W = normal weight on the wheel or track

ℓ = footprint length

n = Bekker soil parameter that accounts for the penetration resistance change with depth.

K_c and K_ϕ = soil penetration resistance parameters

b = footprint width

The resistance due to pushing soil ahead of a wheel or track is small unless the sinkage is rather large compared to the wheel diameter. Equations are available for computing values for this type of rolling resistance[1].

The resistance due to mechanical friction losses applies mainly to tracked vehicles. It accounts for energy losses between the drive sprockets or pulleys and the track itself. It is usually determined experimentally. The total rolling resistance is therefore:

$$R_T = R_C = R_D + R_F + R_S$$

R_C = resistance due to soil compaction

R_B = bulldozing resistance

R_F = frictional resistance

R_S = resistance due to slope

12.2.5. Locomotion Energy and Power

The energy required to move a vehicle is a function of the total rolling resistance and the efficiency of the power propulsion system. The energy consumed due to rolling resistance is:

$$E_R = 1.23 \times 10^{-3} \; R_T \; \frac{\text{kw - hr}}{\text{km}}$$

R_T = total rolling resistance in pounds

When efficiencies are included, the equation becomes:

$$E = \frac{1.23 \times 10^{-3}}{\eta} \quad R_T \; \frac{\text{kw-hr}}{\text{km}} \qquad (12.2\text{-}2)$$

η = the overall efficiency of the power locomotion system presented

Locomotion power versus speed data for the MULE are presented in Section 12.3. The system efficiency was assumed constant; the equation for the calculations made is:

$$P = \frac{1.23 \times 10^{-3}}{\eta} \quad (R_C + R_F + W\sin\beta) \; v$$

v = velocity, km/hr

β = terrain slope

R_C was determined using Equation (12.2-1) with the following vehicle and soil parameters:

W = 9705 lbs (earth weight)

ℓ = 168 inches

$$\left.\begin{array}{l} K_C = 0 \\ K_\phi = 2 \text{ lbs/in}^3 \end{array}\right\} \begin{array}{l}\text{Bekker soil penetration} \\ \text{values for lunar soil}\end{array}$$

n = 1.0

b = 24 inches

It was found to be very small compared with resistance due to friction.

R_F was calculated from the equation:

$$R_F = 0.04\ W$$

The coefficient 0.04 was obtained from Bekker[1] (Page 569, rubber bushed track).

12.2.6. Braking

A simplified equation for obtaining the stopping distance of an off-the-road vehicle is:

$$L = \frac{0.50 \left(\frac{W}{g}\right) v^2}{(AC + W \cos \beta \tan \phi + W \sin \beta)}$$

where

g = acceleration due to gravity

W = vehicle weight

β = terrain slope

C = soil cohesion, psi

ϕ = soil internal friction angle

A = vehicle footprint area

Stopping distance versus speed curves for various terrain slopes for the MULE vehicle are shown in Figure 12.2-1. The curves show that as the slope approaches 30° the stopping distance increases rapidly. This is due to the fact that the vehicle weight down slope cancels out the frictional force developed due to vehicle weight. This shows that it is important to have a large footprint area A if a vehicle is to stop on slopes that approach the soil internal friction angle.

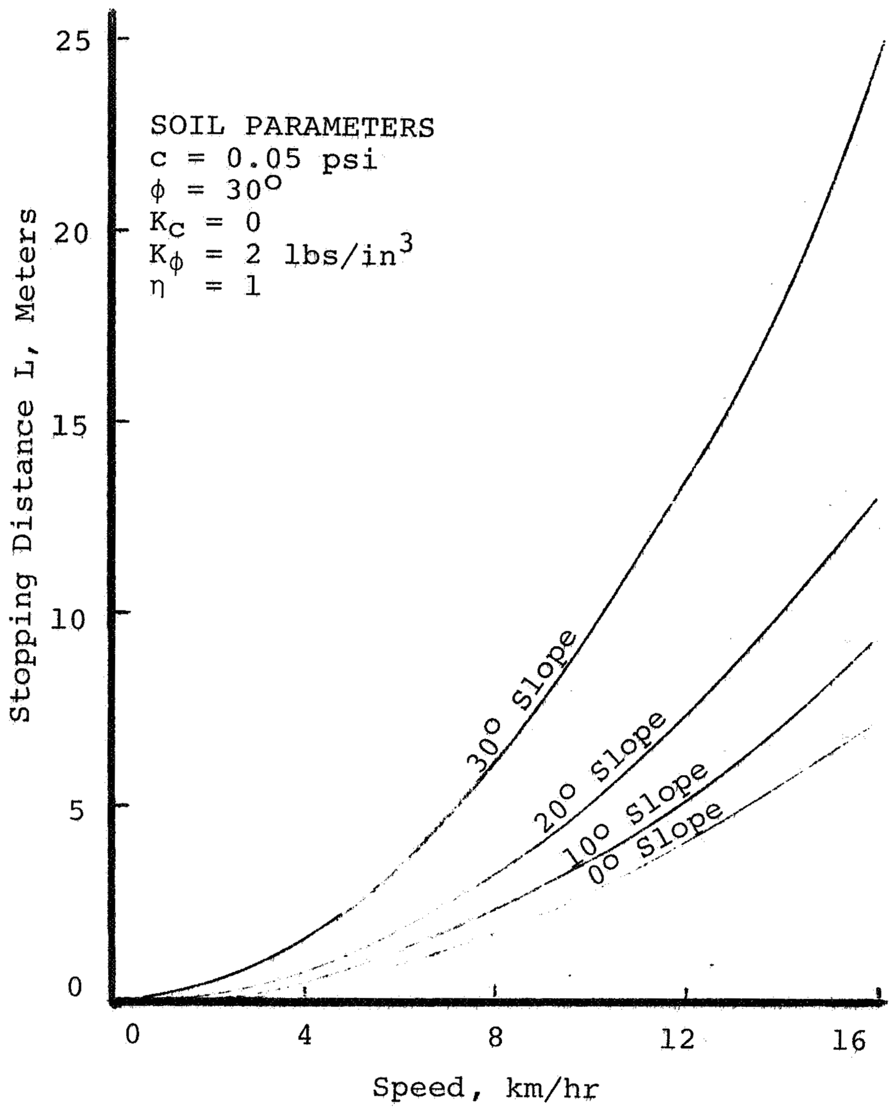


FIGURE 12.2-1 STOPPING DISTANCE vs SPEED FOR THE MULE

12.2.7. Vehicle Slope Climbing Ability

A simplified equation which indicates the pull capability of a vehicle on a slope is:

$$DP = AC + W \cos \beta \tan \phi - R_T - W \sin \beta$$

As the slope approaches the soil internal friction angle ϕ, the equation simplifies to:

$$DP = AC - R_T$$

since

$$W \cos \beta \tan \phi = W \sin \beta$$

$$\text{as } \beta \to \phi$$

This again points out the advantage of having a large footprint area A. This was a very significant reason for selecting the tracked locomotion system rather than the wheel.

12.2.8. Vehicle Stability, Controllability, and Ride Comfort

These parameters largely determine the maximum speed that an off-the-road vehicle can safely traverse a given terrain. It is very difficult to determine specific indices for these parameters since a vehicle which performs well on a certain terrain may perform poorly on another. Simulation methods have been used to determine the performance of off-the-road vehicles in regard to the above parameters[1]. The vehicle parameters and adequate information regarding the terrain to be traversed are considered in the analysis (slope distribution, surface power spectral density, etc.).

Due to the inherently poor riding characteristics of the track-type vehicle, a special type of suspension system was suggested for the MULE. It is described in Section 12.3. The individually suspended, flexible rim track wheels should improve the ride characteristics and stability tremendously compared with a rigid suspension. It should also improve the controllability of the vehicle since the time that vehicle wheels are in the air is a measure of vehicle controllability.[1]

12.2.9. Obstacle Negotiation and Dodging

Large craters and rocks must be bypassed to (1) avoid high probabilities of vehicle immobilization, (2) to keep the vehicle speed as high as possible, and (3) to maintain a reasonable degree of ride comfort for the vehicle occupants.

If all objects in a traverse path are large enough so that they appear on survey maps, routes can be selected instead of considering a random route. Since the lunar surface has millions of small craters and rocks that are not shown by maps, map routing can only serve to minimize problems in regard to the larger craters and rocks.

Researchers have suggested that the distribution of obstacles can be described by statistical processes. If Poisson distributions are assumed, the relationships between spatial densities of individual obstacles and the distribution of distances between them can be determined[1]. The mean distance S_m is

$$S_m = \frac{0.5}{\sqrt{N/A}}$$

where

N = number of obstacles within the area A.

This procedure allows the researcher to determine the free path width that passes through the obstacle field. The values for minimum free path width do not, however, completely define the vehicle width because of vehicle encroachment when turns are made. Scale model investigations are perhaps the best method of determining the vehicle maximum width for vehicle passage for a given obstacle density[1]. Bekker states that vehicle passage is not a difficult problem until craters cover 40 percent of the total area. This analysis, however, appears to neglect the problems due to rocks that are superimposed over the area A.

It is a foregone conclusion that an off-the-road vehicle will have to negotiate or pass over smaller rocks and craters. There is a great variety of forms and sizes of natural terrain obstacles which may stall a vehicle. When the problem is analyzed, however, only a few obstacle configurations cause stalling. There are two basic types of clearance failure modes (CFM); these are hang-up failures and nose-in failures. These are shown in Figure 12.2-2 for convex and concave obstacles.

It can be noted that the MULE rates very well in regard to clearance failure probability. Hang-up failures are a slight possibility due to the track locomotion system. Nose-in failure probability is low due to the large track wheels and the inclined section at the front.

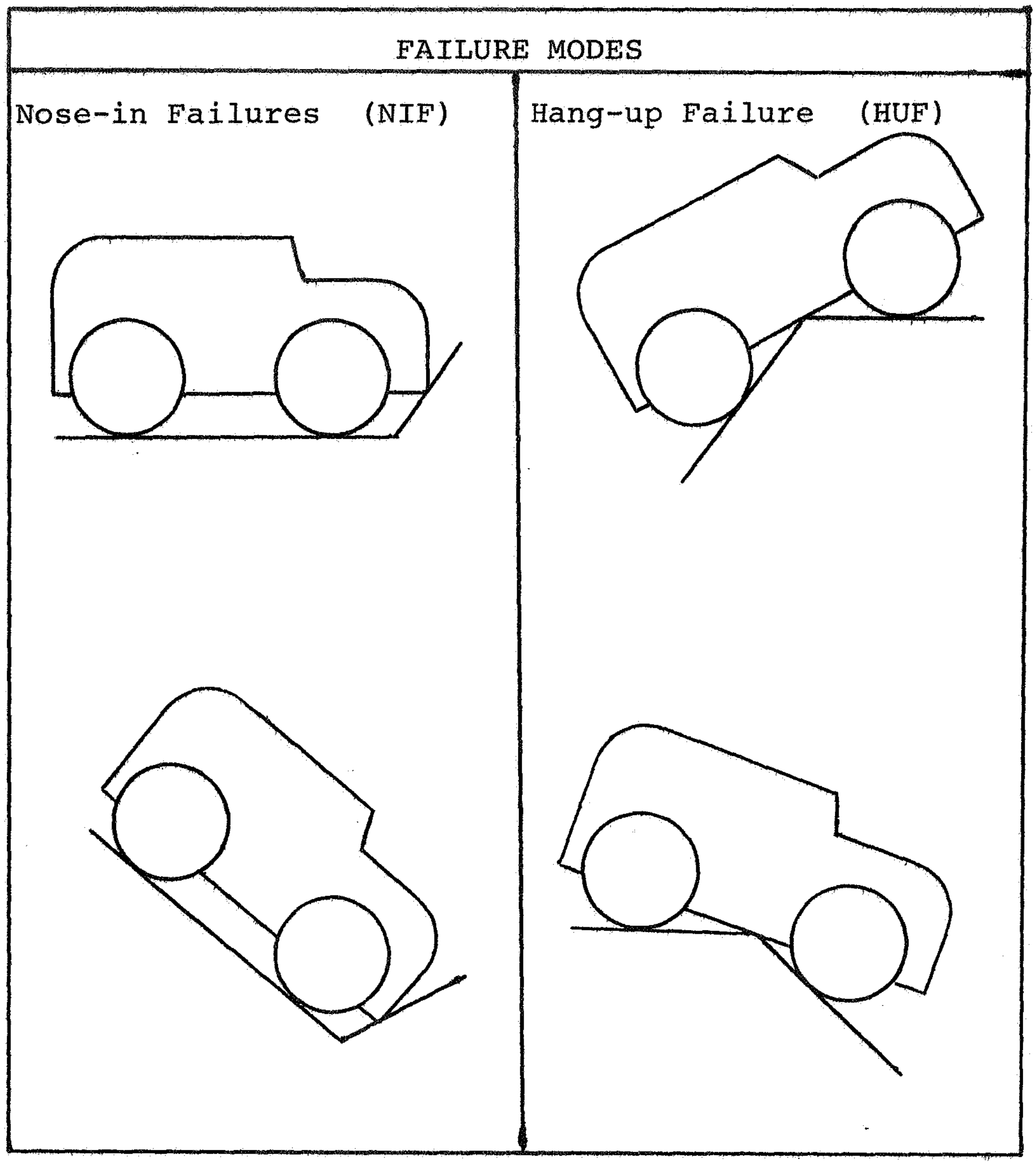


FIGURE 12.2.2. HANG-UP VEHICLE FAILURE MODES

REFERENCE

1. Bekker, M. G., Introduction to Terrain-Vehicle Systems, University of Michigan Press, Ann Arbor, Michigan, 1969

12.3 Locomotive Control

Miltiadis Leptourgos

It was found that tracks were the best means of locomotion and that wheels were second. This conclusion was reached after taking into account many considerations, such as: draw-bar pull capability, slope negotiation, turning radius, and steering. The vehicle has the option of being translated by two means: One is via tracks, the other one via wheels. At any given time, the tracks can be removed allowing the vehicle to run on its wheels.

The average speed for the manned version is 10 kilometers per hour; (kph) while for the unmanned it is between 1 KPH and 2KPH. This is true for both the tracked and the wheeled vehicles. The minimum turning radius for both types of vehicles is 0 feet. This is accomplished by having one set of wheels or track on one side turn in the forward direction, while the other set of wheels or track on the opposite side, turns in the aft direction.

Each wheel is driven individually by an electric motor which is mounted concentrically around the axle of each 4-foot diameter wheel. (See Figure 12.3-1) Braking the vehicle is accomplished by reversing the current flow to the electric motor.

The MULE has four 4-foot diameter wheels on each side. In addition there is a 2-foot diameter wheel in the forward position on each side of the vehicle. (See Figure 11.4-1) The 4-foot diameter wheels are supported in such a fashion that a 6 inch vertical movement, up or down, is allowed. Suspension of each

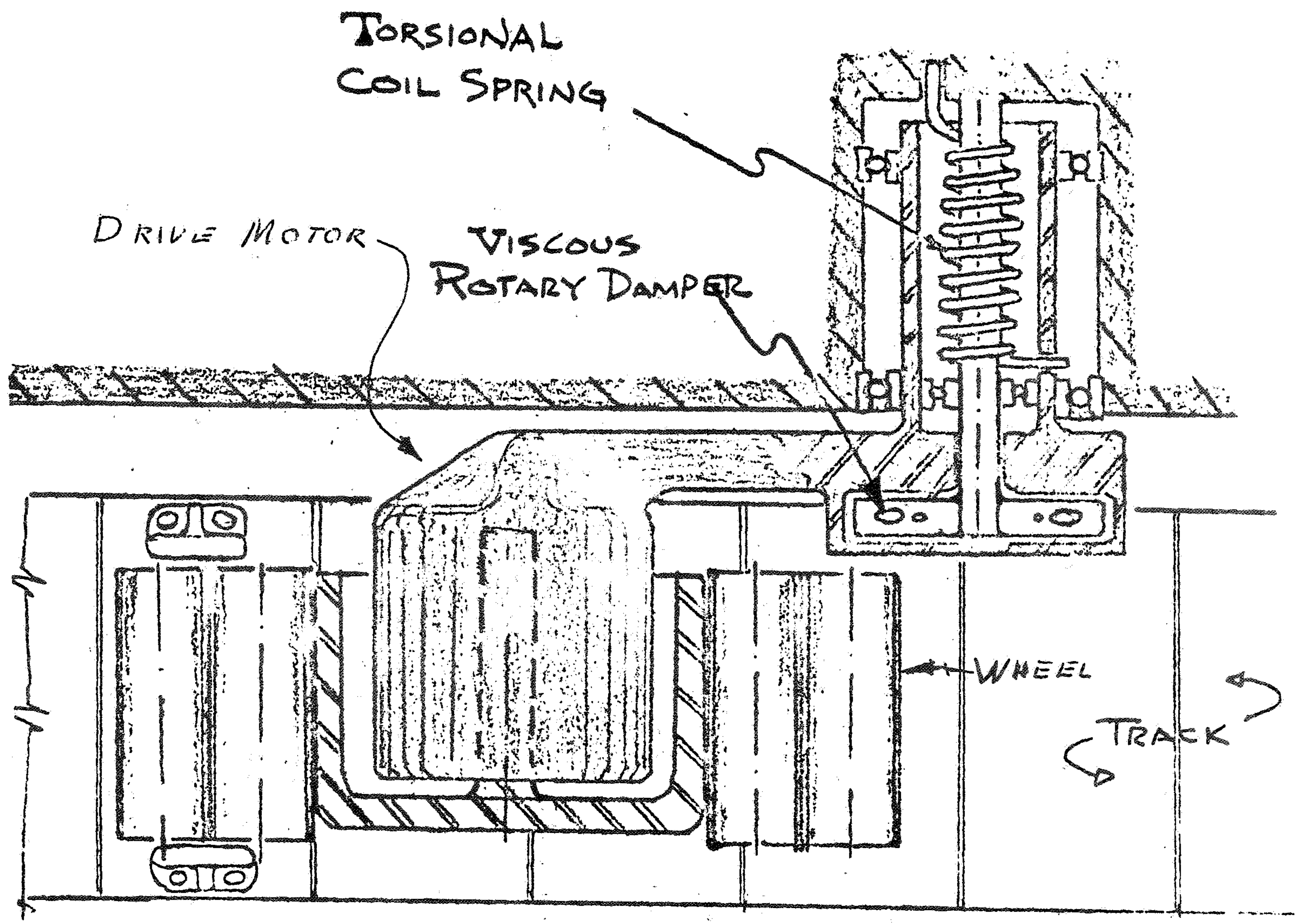


FIGURE 12.3-1. DRIVE AND SUSPENSION SYSTEMS FOR MULE

wheel is accomplished via a S-Type bracket which, on one side, connects to the wheel axle, while the other side slides in a piston type container. The same container serves as a shock-absorber. The latter is of the spring-fluid type, Figure 12.3-1.

There is one track per side. The track is 2-feet wide and approximately 43 feet in total length. The track itself is a continuous belt, which on the outside has cleats. On the inside, metal brackets fastened to the belt, guide the track to keep it on the wheels. The smaller 2-foot diameter wheels are adjustable, and are used to regulate the tension in the track.

12.4. Structure

Miltiadis Leptourgos

Research has indicated that a flexible wheel is desirable. The wheels proposed for the MULE are made from an aluminum-titanium alloy. Each wheel in itself is made from smaller wheels, all of which are surrounded by a final wheel of a 4-foot diameter. This is illustrated in Figure 11.4-1. Consequently, each wheel exhibits a high strength to weight ratio and at the same time is flexible enough to accommodate rough terrain, pressure differences, shock, vibration, etc.

More research and development is needed to produce the desired track. It is suggested, that the track be designed to withstand wear and tear, and pressure. To this end, it should be both flexible and durable. Some of the proposed materials to accomplish this are: nylon, teflon, possibly reinforced with steel wire to produce an integrated-composite track.

12.5. Locomotion Power

J. M. Ulrich
P. S. Shieh

Using the power equation and rolling resistance data developed in Section 12.2, the total locomotion energy can be calculated. From Equation 12.2-2, we see that the net tractive efficiency must be determined. It may be considered a product of the motor, fuel cell, and tractive efficiencies. Thus,

$$\eta = \eta_m \times \eta_{fc} \times \eta_{tr} \qquad (12.5\text{-}1)$$

$$= .9 \times .85 \times .8 = 0.61$$

Therefore,

$$E_t = .002\ R_t\ \text{KWH/KM} \qquad (12.5\text{-}2)$$

For the unmanned mission, the average design speed is 5 KMH and for the manned mission is 10 KMH. (remote control difficulties probably limits the vehicle to 2 KMH in the unmanned mode) For both cases, the minimum speed capability going up a 30^{o} slope shall be 1.0 KMH. Considering first the unmanned mission, it will be further stipulated that 90% of the 1500 KM mission will be carried out at an average speed of 5 KMH (for purposes of maximum energy capability) and 10% at 1 KMH. The mission locomotion time then is:

1350 KM @ 5 KMH = 270 HRS
150 KM @ 1 KMH = 150 HRS
420 HRS

From Section 4.4., we obtain the data for Table 12.5-1.

TABLE 12.5-1 LUNAR SURFACE CONDITIONS

TERRAIN	% of TRAVERSE	SLOPE^{o}
Smooth Mare	90	<3
	98	<6
Rough Mare	80	<3
	90	<6
Hummocky Uplands	50	<3
	80	<6
Rough Uplands	40	<3
	70	<6

In this table, we observe a wide variation in the percentage of traverse distance at the two slope angles. Without a mission profile, an assumption must be made as to an average slope throughout the distance. Since the slope is 6^{o} or less 70% of the time even in the rough uplands, and considering the face that half of these slopes are downward, an average constant upward slope of 3^{o} will be estimated. After the power required for this is determined, a check will be made to assure peak capacity in negotiating a 30^{o} climb.

The rolling resistance values for the variable slopes may now be substituted in Equation 12.5-2 to develop data for Table 12.5-2.

The power required will now be determined.

$$P = E_t(tot) \; x_v \qquad (12.5\text{-}3)$$

where P = Power, KW

v = Velocity, KMH

3° Slope: P = .310 x 5 = 1.55 KW (Avg.)

6° Slope: P = .504 x 5 = 2.52 KW (Peak)

30° Slope: P = 1.485 x 1.0 = 1.485 KW

TABLE 12.5-2. SPECIFIC TRACTIVE ENERGY VS. SLOPE ANGLE

Slope (degrees)	0	3	6	30
E_t KWH/KM	.100	.230	.374	1.35
Damping (15%)	.015	.034	.056	----
Steering (10%)	.010	.023	.037	.135
Accel. (10%)	.010	.023	.037	----
TOTAL, KWH/KM	.135	.310	.504	1.485

Thus, we see that peak capacity must be 2.52 KW and average 1.55 KW. The 30° slope can easily be negotiated at reduced speed.

For the manned mission, the time required to traverse 250 KM at an average speed of 10 KMH is 25 hours. Again assuming an average slope of 3°, and limiting the 6° slope velocity to 5 KMH, we find:

3° Slope: P = .310 x 10 = 3.10 KW (Peak)

6° Slope: P = .504 x 5 = 2.52 KW

30° Slope: P = 1.485 x 1.0 = 1.485 KW

Thus the manned version requires more locomotive power due to the higher average velocity. Of course, the unmanned mission requires the most total energy because of the long time involved. The power system design was, therefore, based on the unmanned mission requirements.

CHAPTER 13

CABIN

J. T. Emanuel

Manned missions of more than 6-8 hours necessitate the inclusion of a closed pressurized cabin for astronaut utilization. Although this cabin is not required to be a "hard" metal cabin, it should be sufficient size to permit suit removal or at least depressurization. The reasons for such a requirement are three-fold. First, the capability of the PLSS to support life during EVA is approximately 6 hours. Second, the BTU production increases considerably for a man in a pressurized suit. Six to eight hours appears to be a practical limit for this increased level of energy consumption. Third, although the Apollo suit provides for human waste disposal, use of such facilities is highly undesirable and should be used only for emergency situations.

To adequately support human life on a 36 hour mission the vehicle cabin should permit the astronauts to operate part or all of the time in a shirtsleeve or depressurized suit environment. Cabin thermal control and life support systems are required to actualize such an environment. Other factors to be considered in a cabin system are ways the operator can affect control of the vehicle, methods of performing extravehicular tasks while allowing the astronaut to remain inside the vehicle, and some alternates to the "hard" cabin concept.

13.1 Cabin Thermal Control

Charles H. Byers

Aside from the basic provisions for metabolism of the man associated with the missions of the MULE, perhaps the most important single control function is that of temperature. Nominally it is desirable to maintain the cabin temperature in the region of $70^{\circ}F$ to $75^{\circ}F$. However, temperatures ranging from $60^{\circ}F$ to $90^{\circ}F$ can be tolerated for substantial periods of time in the event of a failure or partial loss of thermal control. On the other hand the lunar thermal environment is a hostile one, with the surface temperature being as high as $210^{\circ}F$ at lunar noon and as low as $-260^{\circ}F$ during lunar night. The thermal control loop for the cabin has the task of tempering this environment.

A crude drawing of the basic features of the thermal control loop is given in Figure 13.1-1. The coolant, an aqueous glycol solution, is recirculated through a system which provides heat sources and heat sinks which are used as the occasion arises to either cool or heat the cabin. For instance, when cooling is required, the coolant is pumped through the space radiators which radiate heat into space. If additional cooling is required heat is removed from the coolant by the boiling of water which is a product of the fuel cells. Forced ventilation heaters are provided for both the main cabin and the airlock. In addition, the feed oxygen to the cabin, having originated from the cryogenic tanks, must be heated before it reaches the cabin. Finally, there are facilities for the coolant loop to be directly connected into the pressurized suits of both of the cabin occupants. This is provided as a safety measure

TABLE 13.2-1. LIFE SUPPORT SUBSYSTEMS

13.2 Life Support

13.2.1 Crew Station	13.2.2 Atmosphere Control	13.2.3 Food Management	13.2.4 Water Management	13.2.5 Waste Management	13.2.6 EVA & PLSS
Crew Accommodation	Temperature Control	Storage	Storage	Collection	Spare Spacesuit Storage
Instrument Control & Display	Pressure Control	Distribution	Distribution	Storage	PLSS Storage
Personal Equipment Storage	Humidity Control			Disposal	
Health & Safety Equipment Storage	CO_2, Odor, & Contaminant Removal				
Lighting	Air Circulation				
	Oxygen Storage				

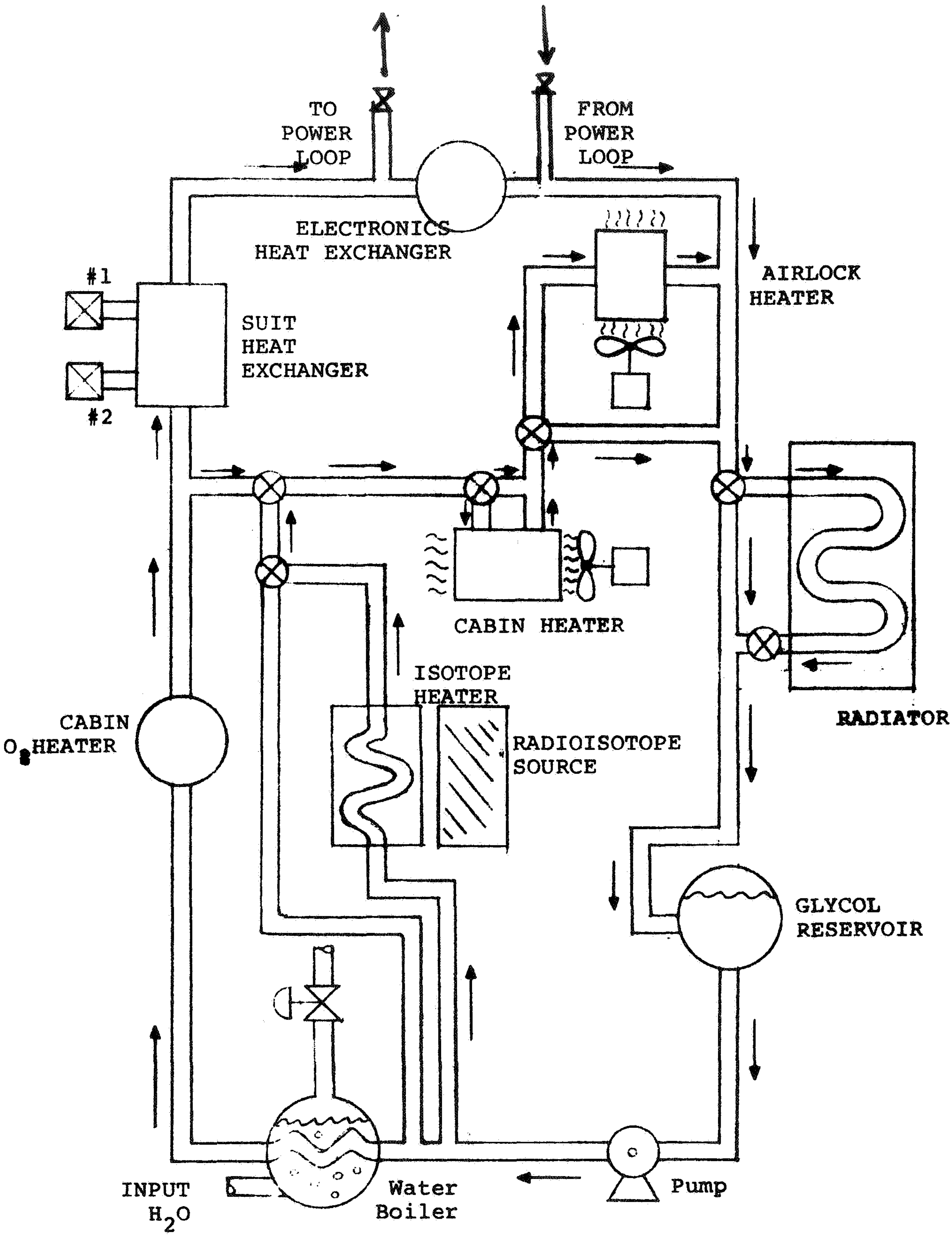


FIGURE 13.1-1 CABIN THERMAL CONTROL LOOP

in the event that a cabin malfunction forces the men to drive the vehicle for long periods of time in pressurized suits. Obviously the other facets of life support are similarly provided for in this emergency configuration.

The astrionics components which are particularly demanding of thermal control are provided with their thermal control capability, as are the critical parts of the manipulator subsystems. While operation in the manned mode will probably not require that these facilities be used extensively, in that these elements will be contained within the cabin and will, therefore, benefit from its uniform temperature, there are possibly some heat producing elements which will require special cooling facilities. On the other hand, during unmanned missions there is no reason to maintain an atmosphere within the cabin, and therefore there can be no thermal control of the entire cabin during such missions. The temperature of the parts of the cabin which contain the electronics must be controlled in order to assure their proper functioning. It is recommended that these portions of the system be situated as much as possible in one confined area and that the details of the coolant loop be such that the thermal control be exercised by conduction. The possibility of the use of radio-isotope heaters imbedded in the electronics package is worthy of investigation.

Finally, it should be pointed out that passive means of thermal control in the areas of the window bubbles are discussed in Section 13.5. The entire question of thermal control is considered in detail in Chapter 21, and therefore has been given only cursory attention here.

13.2 Life Support

H. Y. Chang

Ideally, it is essential to obtain conditions within a spacecraft resembling life on earth so as to protect man against hostile surroundings and to provide for him a hospitable environment. However, because of the high cost associated with space exploration, weight, volume, and power considerations will be the major engineering constraints. Open systems, in which expendables are carried, are optimum for brief missions (less than several weeks). It becomes advantageous to incorporate into the life support system the processing of wastes to regenerate water and oxygen, if the mission extends beyond several months. When contemplating long-term missions (over one year), food regeneration must be considered. The subject of providing life support for extended space flights has been attracting the interest of many research workers. A large amount of literature is available as reflected in the references of this Section.

A typical life support system for short missions may consist of a combination of subsystems as given in Table 13.2-1.

In designing the life support system crew safety , fitness, and well being are guidelines. A proper balance must be maintained between periods of work, relaxation, and sleep. From the comfort standpoint, a shirtsleeve environment is essential; and the astronaut will be most efficient if he is comfortable. Conceivably space operations will necessarily involve extravehicular activities (EVA), such as outside maintenance of the space vehicle and exploration on the planet's surface. In achieving these objectives, the EVA system will use a spacesuit/backpack portable life support system (PLSS) combination.

The size of a life support system is influenced by crew metabolic requirements, which will have to be adjusted for the combined effects of crew physical activity levels, reduced gravity on the lunar surface, and artificial atmosphere inside the spacecraft. The design parameter for physiological tolerance, metabolic rates under shirt-sleeve environments, and performance requirements for PLSS are given in Tables 13.2-2, 13.2-3, and 13.2-4. A material balance diagram is given in Figure 13.2-1.

TABLE 13.2-2 DESIGN PARAMETER FOR PHYSIOLOGICAL TOLERANCE

Parameter	Design Criteria
Temperature, ^{o}F	70-80
Oxygen partial pressure, psia	3.7-5.0
Carbon dioxide partial pressure, mm Hg	0-8
Relative humidity, %	50 ± 20*

*This specification is the most commonly used, though it is both ambigous and imprecise. We can say that the partial pressure of water vapor should be kept at 5-15 mm Hg, which is an improved expression. A much better, yet more complicated, specification is the use of a "comfort index", which interrelates the various effects of relative humidity, temperature, partial-g, reduced pressure, and air composition [5,6,10,17].

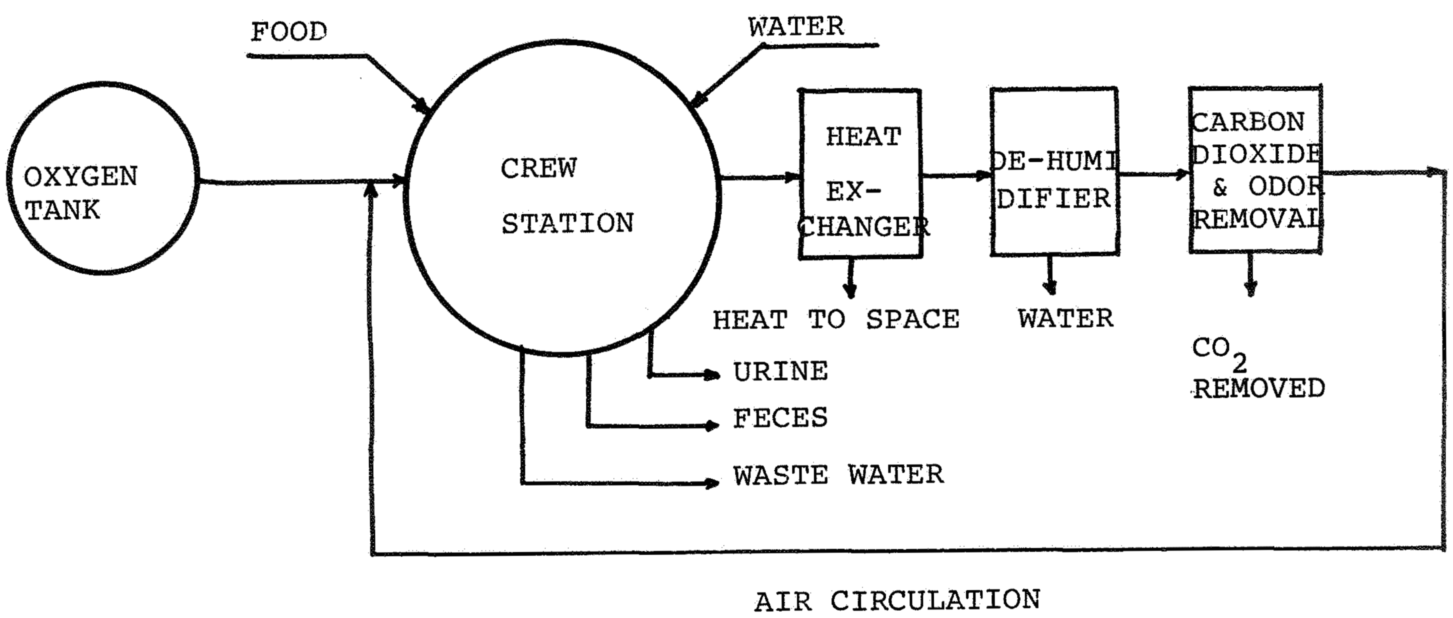
TABLE 13.2-3. METABOLIC RATES UNDER SHIRT-SLEEVE ENVIRONMENTS

Metabolic requirements		Design Criteria lb/man-day
Oxygen		2.0
Drinking water (including food reconstitution)		8.0
Hygiene water		3.5
Dehydrated food		1.5
	Total	15.0
Waste production		
Carbon dioxide		2.25
Water vapor (perspiration and respiration)		5.4
Waste wash water		3.5
Urine		3.5
Feces		0.35
	Total	15.0
Metabolic heat	12,000 BTU/man-day	

TABLE 13.2-4. PERFORMANCE REQUIREMENTS FOR PLSS

Mission duration (maximum), hr	6
Average metabolic rate, BTU/hr	2000*
Maximum metabolic rate, BTU/hr	3500*
Contingency duration, hr	0.5
Maximum external environmental heat loads	
Daytime heat gain, BTU/hr	250
Nighttime heat loss, BTU/hr	-350

*The very high metabolic rates for EVA listed here are based on data of Prince, Iles, and O'Reilly [16]. The effects of pressurized spacesuit and reduced gravity on hard work on lunar surface have been taken into consideration.



Input	lb/man-day	Output	lb/man-day
Oxygen	2.0	Carbon dioxide	2.25
Metabolic water	8.0	Water vapor (perspiration & respiration)	5.4
Hygiene water	3.5	Waste wash water	3.5
Food	1.5	Urine	3.5
		Feces	0.35
TOTAL	15.0	TOTAL	15.0

FIGURE 13.2-1. LIFE SUPPORT SYSTEM (CABIN) MATERIAL BALANCE

13.2.1 Crew Station

The crew station should provide space and facilities for such activities as cabin egress and ingress, eating, sleeping, waste disposal, donning and doffing of pressure suits- equipment check out, and speciman storage. Windows are placed at convenient locations to permit direct visual observations. Lighting should be provided with proper illumination and sufficient contrast. Instrument and control displays are required for operation and monitoring systems. They are functionally grouped and arranged, and should be provided with manual override so that maximum reliability can be assured. First aid and personal hygiene supplies as well as EVA equipment should be included.

13.2.2 Atmosphere Control

The atmosphere control subsystem has the complex function of providing oxygen storage, pressure control, temperature control, humidity control, and carbon dioxide, odor, and contaminant removal. It is common practice to use a pure oxygen atmosphere at reduced pressure for short missions. The system is relatively simple and light in weight, and has proven reliability. However, there is concern for using pure oxygen for prolonged periods[4,11,14,15]. The recommended procedure is to use sea level air for long missions, if practical. Otherwise, an oxygen-nitrogen or oxygen-inert gas mixture should be employed instead.

Oxygen storage should cover the needs for crew metabolic requirements, cabin pressurization, and leakage losses. The design criteria for metabolic requirements has been given in Table 13.2-3. Pressurization and leakage losses are dependent upon cabin configuration. The method of oxygen storage may be either high pressure, cryogenic, or solid peroxides. For short missions high pressure storage has the advantage of simplicity and light weight.

Carbon dioxide removal is effected either by chemical absorption or by physical adsorption. Chemical absorption on lithium hydroxide is the best available method for short missions. The reaction can be represented by the following equation:

$$2\ LiOH + CO_2 = Li_2CO_3 + H_2O$$

This is an exothermic reaction. The heat evolved is 875 BTU per pound of carbon dioxide absorbed, with the reaction product water in the vapor state. The carbon dioxide concentration level can be controlled by varying the air flow rate.

Dehumidification is accomplished by a heat exchanger and a separator. The dew point temperature fixes the humidity and this is a straight forward approach. Three different sources contribute to the heat load imposed on the cabin -- crew metabolic heat, equipment heat generation, and external heat flux. In order to maintain thermal equilibrium, the control system is composed of a heat transfer fluid loop, a space radiator, and a water evaporator. The space radiator provides the primary heat sink. During lunar day, when the external temperature is high, water boiling can be used as heat sink. The steam is vented to space. Such a system combines simplicity and reliability. The disadvantage is that coolant water consumption will reach a prohibitive level for long missions. Then, other more competitive methods must be considered.

The problem of odor and trace contaminants in the atmosphere is inherent to any closed or semiclosed ecological system. They can be controlled within acceptable level by charcoal absorption and particulate filter. For long missions catalytic combustion can be used to remove odor and contaminants. When the contamination level is high, total vent is an effective way of cleaning the atmosphere in a short period of time. The maximum acceptable level and emergency limit for space cabin contaminants have been recommended by the Fourth Annual Conference on Atmospheric Contamination in Confined Spaces, 1968. Some of the data are compiled in Tables 13.2-5 and 13.2-6.

TABLE 13.2-5. RECOMMENDED MAXIMAL ACCEPTANCE CONCENTRATION FOR CONTINUOUS EXPOSURE*

Provisional limits for space cabin contaminants for 90 days

Air contaminant	ppm
n-Butanol	10
2-Butanol	20
Carbon monoxide	15
Chloroform	5
Dichloromethane	25
Dioxane	10
Ethyl acetate	40
Formaldehyde	0.1
2-Methylbutanone	20
Trichlorethylene	10
1,1,2-Trichloro, 1,2,2-trifluoroethane and related congeners	20

*Data taken from Proceedings of the Fourth Annual Conference on Atmospheric Contamination in Confined Spaces, 1968.

TABLE 13.2-6. PROVISIONAL EMERGENCY LIMITS FOR SPACE CABIN CONTAMINANTS*
(for 60 min.)

Air contaminant	ppm
2-Butanone	100
Carbonyl fluoride	25
Ethylene glycol	100
2-Methyl butanone	100
1,1,2-Trichloro,1,2,2-trifluroroethane and related congeners	200

*Data taken from Proceeding of the Fourth Annual Conference on Atmospheric Contamination in Confined Spaces, 1968.

13.2.3 Food Management

The food consumption will be 1.5 lb. of dehydrated food per man per day, composed of approximately 15 percent protein, 25 percent fat, and 60 percent carbohydrate. This is based on a 3000 Kcal per day diet. If excessive EVA is involved, the amount of food intake will have to be raised. The food is reconstituted for consumption by adding water. For long missions, a small supplement of vitamins and minerals as well as amino acids may be necessary. For extremely long missions (more than one year), physio-chemical and biological synthesis of food may be advisable. Such a system will require extensive research and development.

13.2.4 Water Management

As given in Table 13.2-3, eleven and a half pounds of water per man per day must be supplied. It is stored in a portable water tank. This amount does not include the water required in the evaporator of the thermal control system. If fuel cell is used to produce power, water is obtained as the by-product, and the amount necessary to be carried on board can be correspondingly reduced. Fuel cell water is generally considered to be of high quality, since it is chemically pure. However, the taste is disagreeable to some, due to the difficulty in separating hydrogen gas from the water.

Waste water will be collected in a separate tank and can be stored for emergency cooling use. For long missions, a water recycling process will have to be integrated into the system.

13.2.5 Waste Management

The collection, treatment, and disposal of feces, urine, food residues, packaging, and body wastes must be performed in a convenient, sanitary, and inoffensive manner. Bacteria, odors, and decomposition products must be eliminated or contained during storage. The recommended method is to collect and store urine and other liquid wastes for emergency cooling. Fecal matter and other solid wastes are collected in plastic bags and manually mixed with disinfectants for temporary storage on board, with final disposal at resupply.

13.2.6 EVA and PLSS

The astronauts will arrive wearing their spacesuits and backpacks (PLSS). They will also wear a water cooled undergarment.[3,16] Normally when on board, under shirtsleeve environments, the PLSS will be disconnected. The ventilating air stream in the cabin provides metabolic oxygen, pressurization, carbon dioxide, odor, and contaminants removal, and humidity control. The liquid coolant loop provides thermal control. When an astronaut leaves the cabin for EVA, environment control will be provided by spacesuit/backpack combination. The item weights of a PLSS unit (6-hr duration) are given in Table 13.2-7. Extra PLSS units and spare spacesuits may be necessary for backup purposes.

TABLE 13.2-7. WEIGHT OF PLSS*

Item	lb
Basic dry unit, less items separately listed	26.9
Extra structure	1.5
Battery	15.0
LiOH canister	2.2
LiOH charge	5.4
Oxygen bottle	6.0
Oxygen charge	2.0
Water bottle	1.0
Water charge	15.0
Manifold	9.1
TOTAL	84.1

*Data taken from "Preliminary Design Study of a Lunar Local Scientific Survey Module," Final Technical Report, D2-83015-1, AiResearch Manufacturing Division, Los Angeles, Calif., June 1966.

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13.3 Displays and Controls

R. R. Pikul
J. T. Emanuel

In our conceptual design we did not consider the configuration of displays and controls for most subsystems. Room has been provided in the cabin for switches, gauges, and indicators that will be required to activate and control the life support, power, communication, and other subsystems. Considerations were given mainly to methods for controlling the vehicle and the manipulators.

Figure 13.3-1 shows the conceptual design of the forward instrument panel. In the manned mode the primary source of visual information will be direct line of sight through the forward spherical viewing ports (See Section 13.5). The secondary sources of visual information consist of a TV monitor and an obstacle avoidance display. The TV monitor will display information from the forward or rear external cameras as well as information from Earth, the LOSS, or lunar base. The obstacle avoidance monitor will display data to supplement the one meter resolution maps that are used in vehicle navigation.

Basic vehicle motion, i.e., forward, reverse, left, right, will be affected by a two dimensional joystick located on the astronauts chair arm rest. A speed control, allowing a continuous velocity spectrum, will be located adjacent to the joystick. As with most tracked vehicles, for normal operation the speed control will be set at a value corresponding to the maximal speed desired during that traverse. All movement of the vehicle, including forward and reverse speed will then be regulated by use of the joystick. Push the joystick forward and the vehicle moves forward, the farther forward the stick is pushed, the faster the vehicle goes. Joysticks were considered for much of the vehicle control due to their simplified method of operation requiring the use of only one hand.

Manipulator controls will be located on fold-down panels stowed under the control panel when not in use. Section 13.4 describes the manipulator controls in greater detail. Audio communication with extravehicular locations will be via headphones.

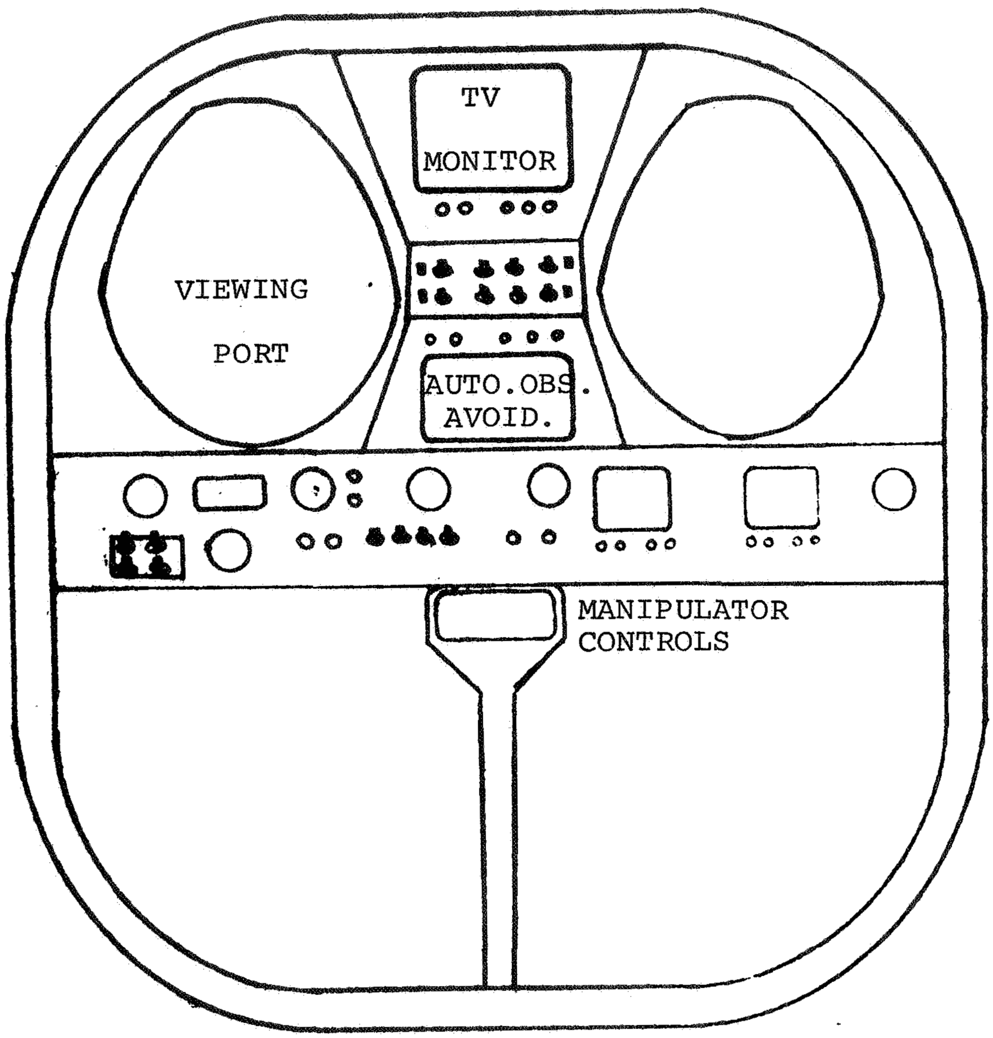


FIGURE 13.3-1 CONCEPTUAL VIEW OF FORWARD OPERATOR CONSOLE

13.4 Manipulators

Richard R. Pikul

Manipulators are a subset of the general grouping labeled "teleoperators". By definition a teleoperator system always contains a man in the control loop and thus can be distinguished from androids and preprogrammed machines. Manipulators should be considered in lunar or extraterrestrial exploration when the following situations arise:

1) hazardous tasks are to be performed or hazardous environments are encountered.

2) endurance greater than that of an extra-vehicular astronaut is required.

3) A savings in weight and cost can be realized without affecting mission goals, i.e., when man can be spared routine missions. For our particular lunar missions, manipulators will be used to collect surface samples and deploy science packages or other cargo in both the manned and unmanned modes of vehicle operation.

There are basically two broad classifications of manipulators, the unilateral and bilateral systems. Unilateral systems are basically open loop in operation without force or motion feedback, but normally with direct or indirect visual feedback. Common examples include cranes, bulldozers and the Surveyor lunar exploration vehicle. Bilateral systems are closed loop with continuous force and position feedback. This latter system is more effective for accurate manipulation, but suffers from weight penalties in the form of force feedback actuators and space penalties in the form of control station master arm working volume. An example is the Argonne National Laboratory Model E-3 Electric Master/Slave Manipulator[1]. The unilateral system is inherently lighter, but for accurate tasks, such as film-cassette-removal and replacement, it expends more time and energy per task due to the complete dependence on visual feedback. This time and energy can vary from 3-10 times that for a bilateral system.[1]

Master to slave arm linkage in either system may be of a mechanical or electrical nature. Mechanical units incorporating tapes, cables and ball joints are lighter in weight and more reliable but present several problems. First, the direct mechanical linkage prevents remote operation beyond a few feet and secondly, the mechanical systems require hull penetrations that will result in sealing problems in a lunar environment. A further problem results in that a normal 1:1 master/slave ratio required from 4-19 cubic feet of working volume for anthropomorphic operator control. Figure 13.4-1 shows a schematic of a typical master/slave ball joint assembly. Electrical linkages allow for remote control and the use of several types of input devices for the same manipulator slave. Panel control can be used at the operator station with a combination of joystick, panel switch and potentiometer control. Other control station inputs can be a set of anthropomorphic master arms or a set of scaled down master arms with both master units incorporating panel switches and potentiometers to augment natural operator movements. An example would be the inclusion of 360° wrist rotation capability and arm extension.

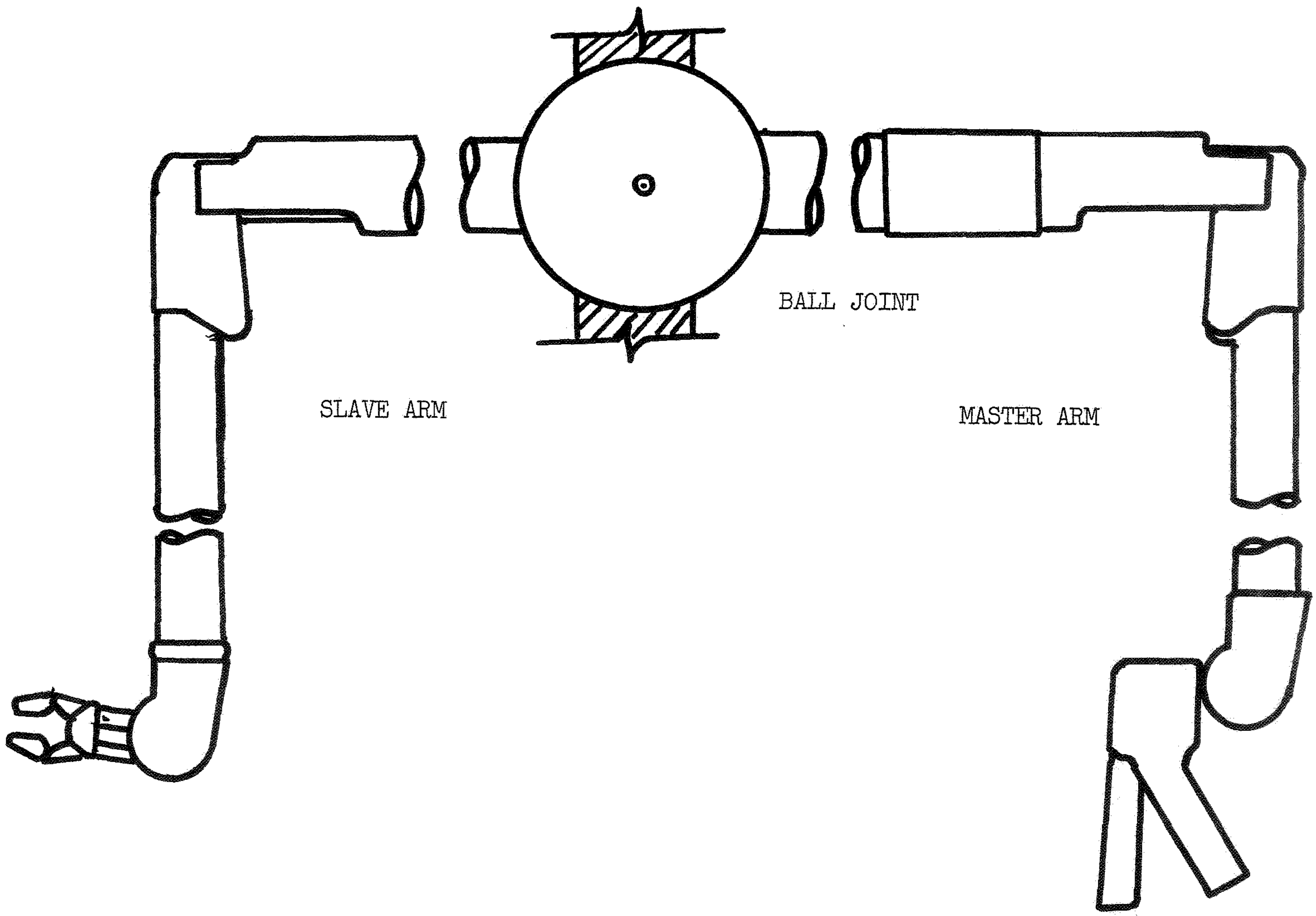


FIGURE 13.4-1 SCHEMATIC OF TYPICAL MASTER/SLAVE BALL JOINT MANIPULATOR

Anthropomorphic and scaled master units both use proportional drives to control the slave unit. Slave arms can be electro-hydraulic, electric motor or hydraulically operated as well as mechanically, using tapes and belts, but only the electric motor operation is feasible for lunar operation.

When the effects of time delay are added into the concept of remote manipulator control both unilateral and bilateral systems degenerate from continuous operation to a "move and wait" strategy with discrete movements by the operator whenever feedback delays are greater than 0.3 - 0.5 seconds[2]. Considering time delays that will be encountered from likely remote stations (see Section 15.3 Remote Control) the force and position feedback advantages of the bilateral system become less important in overall performance.

For the MULE vehicle concept the following recommendations are made for future detailed design. Since time delays will normally be greater than a half second for remote vehicle control, there appears to be little advantage to using a bilateral system with its inherent weight and master volume penalties since a "move and wait" strategy will be the primary mode of operation. Hence, an electric unilateral system is recommended with television feedback and additional feedback in the form of lights or buzzers which are actuated on contact and when force or reach limitations are exceeded. Since the operations to be performed by the manipulators are rather well defined the sophistication of a dual armed slave or even of a six degree of freedom arm is not deemed necessary if the science packages are properly designed. For the surface sampling operation a single forward six degree of freedom manipulator arm is recommended with a scoop end. Since the science packages will be stowed at the rear of the MULE a jib crane manipulator is recommended that will be capable of rotating 360° about the mast and have a movable hoist located on the arm. A steel cable and hook arrangement should be sufficient for package deployment if a mating ring is provided on the package structure and self leveling and preprogrammed initialization are designed into the package. Figure 13.4-2 shows a conceptual view of the recommended manipulators.

In the manned mode, control of both manipulators will be via panel control. For the forward manipulator a six degree of freedom plus gripping motion joystick is recommended with an additional preprogrammed operation to deploy the arm from its stowed position. Figure 13.4-3 shows how such a sophisticated joystick would translate human arm motions into final slave arm movements. Visual feedback for the forward manipulator will be direct line of sight by the operator through the forward viewing ports. Force and arm extension limitations will be noted by panel lights. The rear crane will also be panel controlled in the manned mode but operation will be simplified by switches and potentiometers to control the hook location. Since the operator will not be able to see the crane from within the cabin, visual feedback will be via television cameras located on the top of the cabin and on the hoist of the crane. It should be noted that the cabin camera is part of the vehicle control system and has a pan and tilt capability. At the remote control master station, 1:4 master units are recommended to rapidly rough position the manipulators through the use of propor-

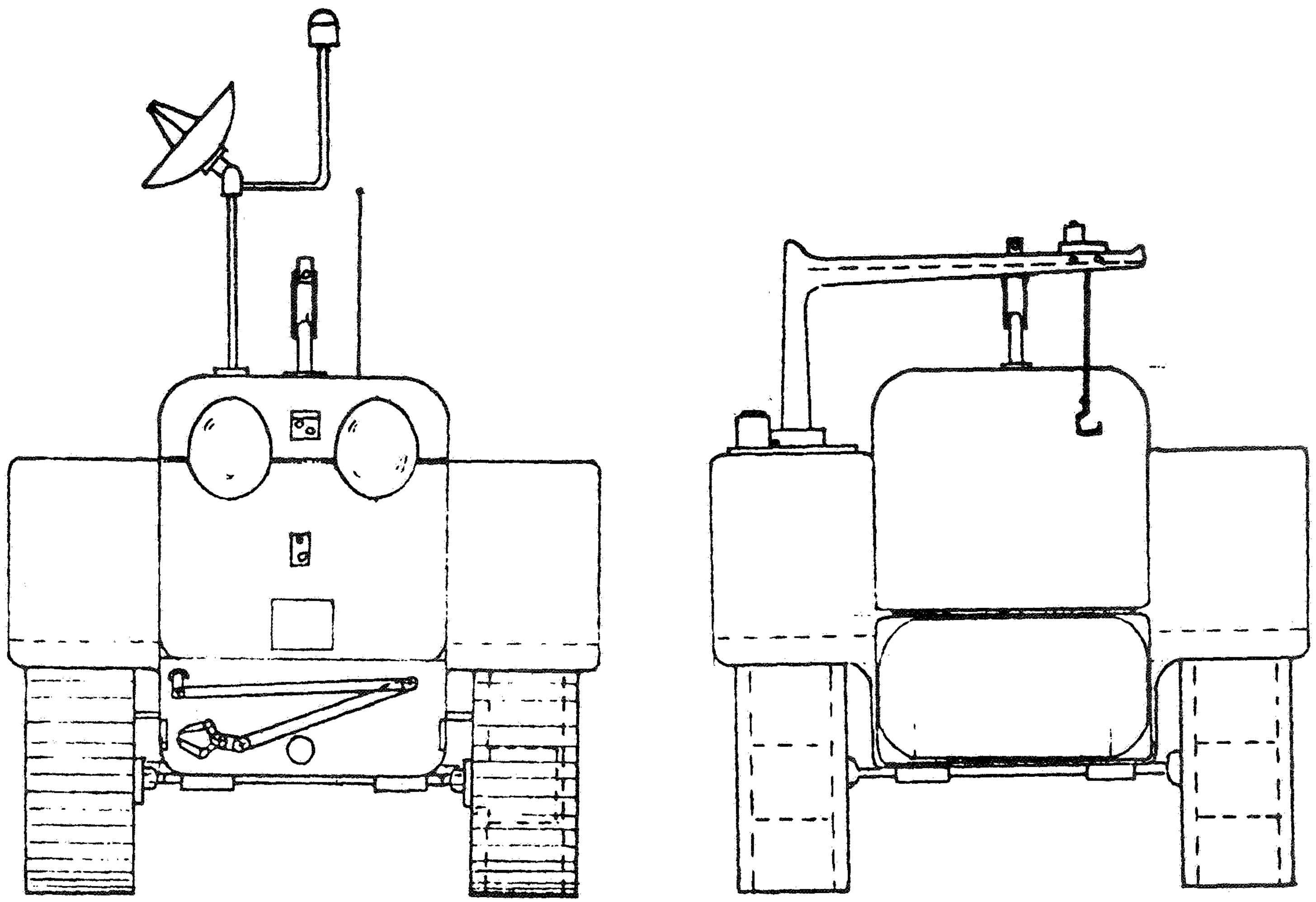


FIGURE 13.4-2 MULE WITH FORWARD AND REAR MANIPULATORS

MOTION TABULATION	
SYMBOL	MOTION
A	SHOULDER ROTATION
B	SHOULDER PIVOT
C	ELBOW PIVOT
D	WRIST ROTATION
E	WRIST PIVOT
F	WRIST DEFLECTION
G	HAND GRIP

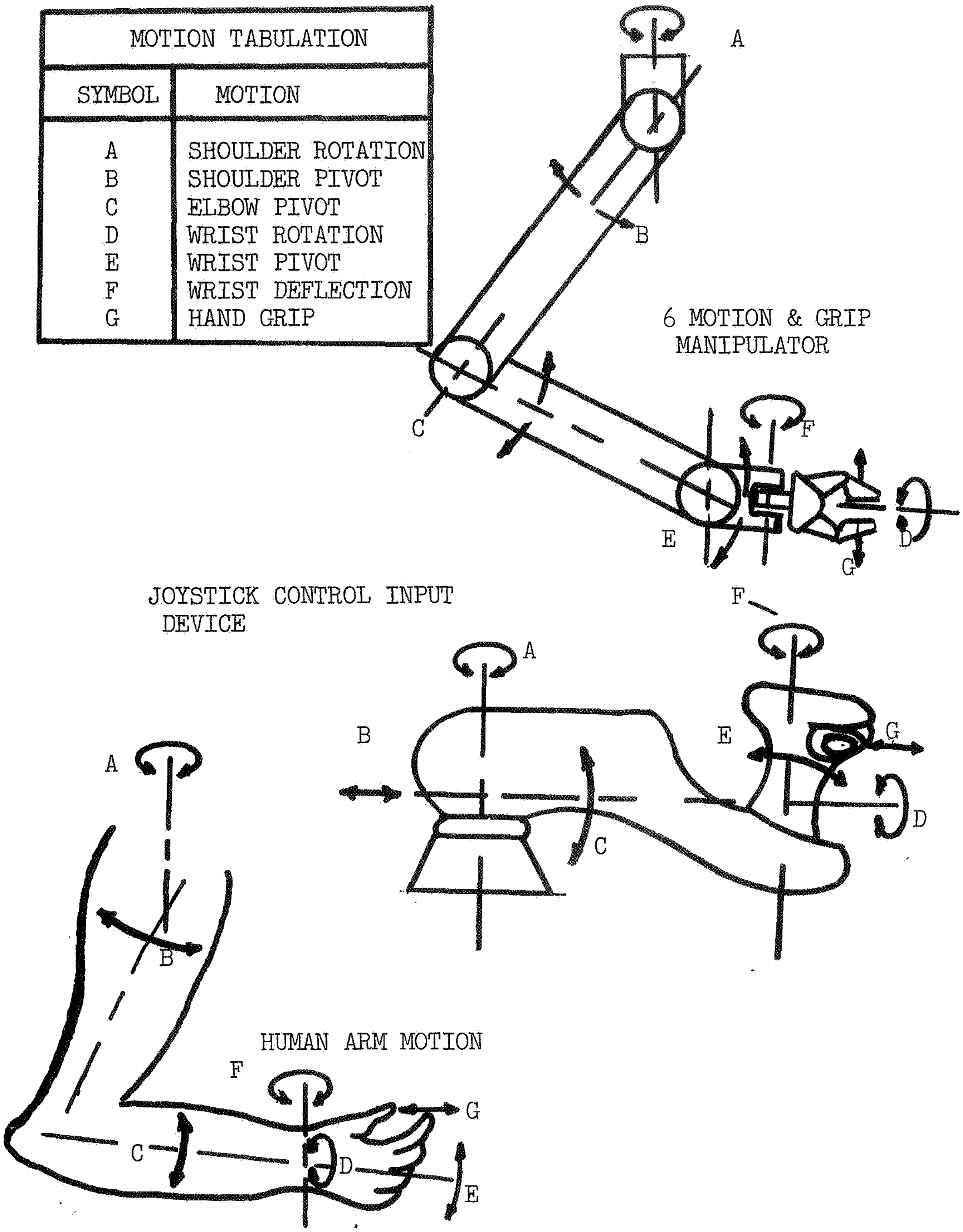


FIGURE 13.4-3 HUMAN ARM & RELATED JOYSTICK AND SLAVE MOTIONS

tional control. Initial preprogrammed deployment is again recommended with limitation feedback similar to that for the manned mode. Refined manipulator movement can be achieved by panel controlled input similar to that used in the MULE cabin. All visual feedback is by necessity indirect via television sensors and displays. Besides the cameras noted for the crane an additional camera is mounted at the front of the vehicle to aid in forward manipulator control.

Since the manipulators required for this vehicle are special purpose in nature, no advantage could be found for elaborate units capable of performing unnecessary operations. The units recommended appear to be the simplest, and hence, probably most reliable units and are for the most part mission and vehicle orientated. Since this study was basically conceptual in nature no detailed calculations were made as to weight and power requirements. Based on other manipulators[1] a weight of 50 pounds for the forward arm and 200 watts power should be sufficient.

The rear crane will probably weigh in the neighborhood of 200 pounds and require approximately 200 watts peak power, exact figures being dependent on the weight of science packages to be deployed.

13.5 Cabin Concepts

Charles H. Byers

During the course of the program several concepts were proposed which could serve as crew shelter during extended missions. This section reviews these ideas as well as the final concept which met the requirements of the program. Also included in this section are descriptions of the facilities for crew comfort, sleeping facilities, and the viewing ports.

13.5.1 Shelter Configurations

The specification of a 36-hour manned sortie for this vehicle led the group to consider several alternative means of providing shelter. First it was considered that the men could spend the entire period in their suits and thus avoid any need for a cabin. The argument in favor of this approach is basically that the improved suits which will be available in the period of the 1980's will possibly allow the man to rest or even sleep while the suit is pressurized, thus realizing a large saving in weight. The arguments against such an approach to the provision of a habitable environment are numerous. First, no satisfactory means of feeding the man in the suit has been devised. Waste management problems remain unsolved. The level of effort required to perform the mission tasks in a pressurized suit is extremely high even while the simplest tasks are being performed. The loss of the pressurized suit through puncture or malfunction of the PLSS leaves the man without any back-up means of sustaining life. Based upon these arguments, it was decided that some form of shelter in addition to the space suit was an absolute necessity.

Several concepts which could possibly provide this function are listed below in the order of increasing weight.

1. The Inflatable Space Tent
2. The Completely-Flexible Cabin
3. The Rigid Cabin for Driving with Flexible Airlock which is expandable to crew sleeping quarters.
4. The Rigid Driving Cabin with reclining seats for sleeping and a small flexible airlock.
5. The Fully-rigid Cabin and Airlock.

Each candidate will be discussed in order and the reasons for the adoption of the fifth candidate will be given in detail.

The concept of the "Lunar Tent" was first proposed by the Goodyear Aerospace Company[1] as a means of extending the stay-time of the Lunar Module on the Moon. An artist's conception of the assembled unit is given in Figure 13.5.1-1. The overall dimensions of the unit are a total length of 17 feet and 7 foot diameter. As is evident from the figure, the unit is completely self-contained. Facilities could be provided which would allow it to utilize the resources of the vehicle to provide the environmental control. The entire structure may be collapsed to fit in a container which is 5 feet square by 2.5 feet high. Again, it is evident from the drawing that the unit must be assembled while the men are fully-suited, thus implying a rather lengthy assembly procedure (approximately one hour). The wall construction is of sufficient interest to warrant further discussion. A typical wall cross-

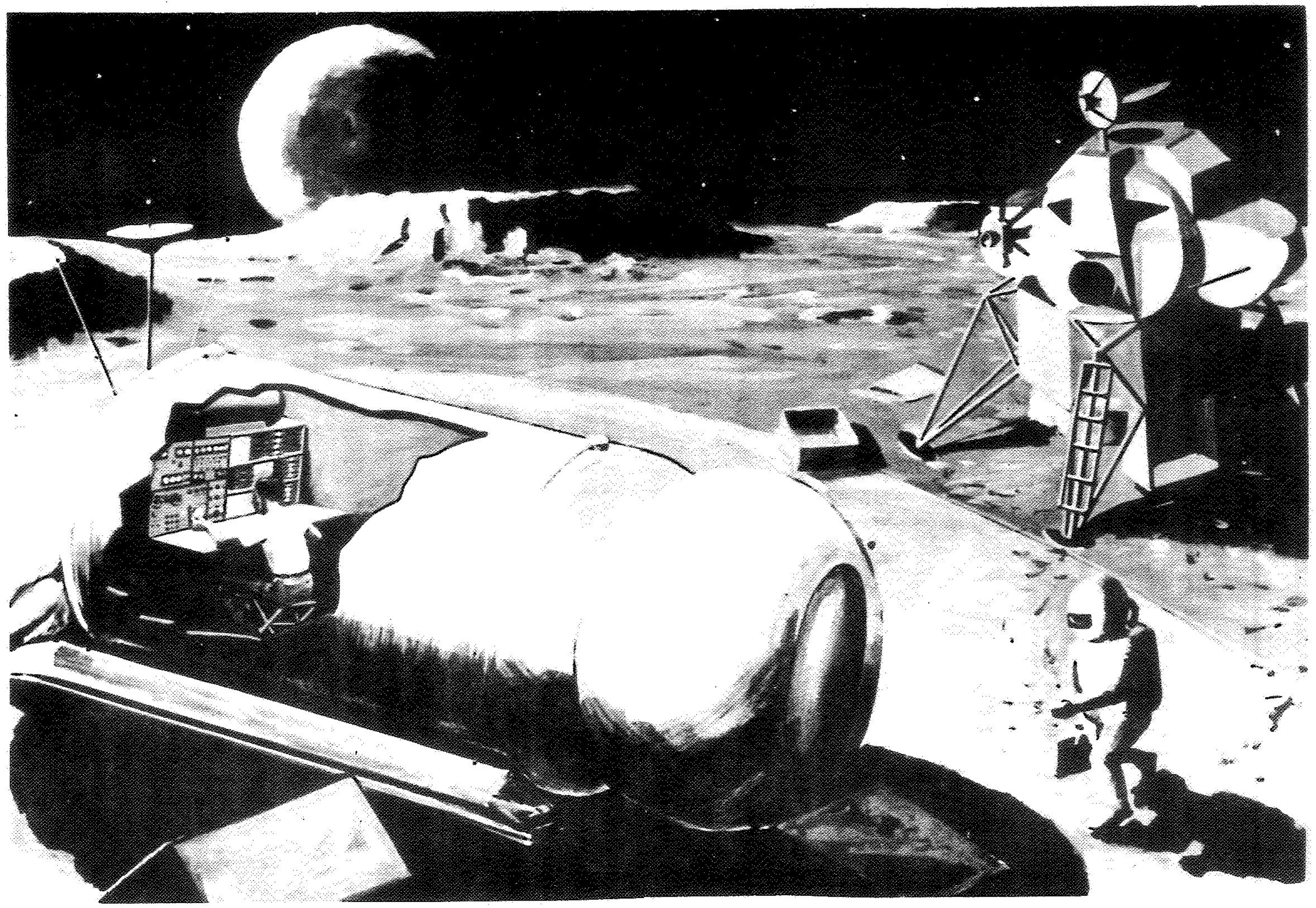
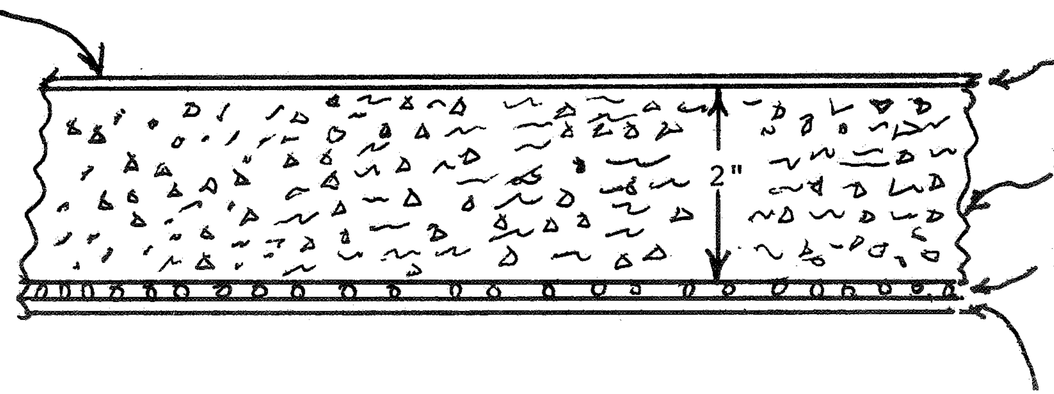


FIGURE 13.5-1 FULLY-ASSEMBLED GOODYEAR LUNAR TENT



+ The outer cover is a cloth laminate consisting of the following layers.

1. Nylon cloth
2. Polyester Adhesive
3. Capran Film
4. Polyester Adhesive
5. Nylon Cloth

* Inner Wall

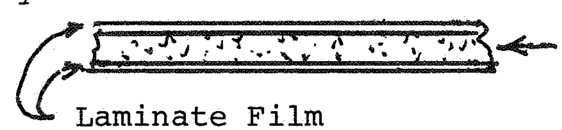


FIGURE 13.5-2 FLEXIBLE TENT WALL CONSTRUCTION

section is shown in Figure 13.5.1-2. The 2-inch thick layer of polyurethane foam which forms the major fraction of the wall's thickness provides protection from meteorites and the various forms of radiation present on the lunar surface in addition to affording protection against heat leaks by conduction. The outer and inner walls are laminates. The former is essentially a cloth cover coated with Zinc oxide paint which has desirable heat radiative properties. The latter is the bladder which contains the 5 psi pressure difference between the interior and the lunar environment. Two novel features of this unit are worthy of mention. The first is a 45 foot in diameter mat which, through its high emissivity, provides for radiation of heat away from the shelter during lunar day. The other is a thermal blanket which may be raised around the tent and which virtually eliminates heat loss from the unit during lunar night. An estimate of the weights involved in the execution of a two-man mission for both the independent and umbilical modes is given in Figure 13.5-3. For the nominal 1.5-to-2 day mission the umbilical mode is favored with a total weight of about 700 lb. It should be noted that this figure includes consumables, but not the difference in weight in the main vehicle caused by the umbilical system. From a safety viewpoint a redundant system allowing both modes to be employed would be best. Bearing in mind the fact that 12-foot-long tent would be adequate in the current study, a 500 pound tent should be a readily realizable design.

The reasons for removing the tent from consideration are numerous. The following are among the more important. As we have already mentioned, the assembly time is quite large. Thus, each time the unit would be required for eating, sleeping or hygiene reasons, a long time period for assembly would be required. The design, as it exists, does not allow for the unit to be stowed, thus, major design changes might be in order. Finally the men must still drive a vehicle which does not have any cabin, and, thus, must be fully-suited. Since some missions call for extended driving times a vehicle without a cabin was eliminated from further consideration.

The completely-flexible cabin concept consisted essentially of modifying the lunar tent to allow it to be used as a structure in which the vehicle could be driven and the other functions be performed. The requirements for a solid structure to which one could attach some of the elements of the design, such as the windows, hatches, and manipulators is a strong point against such a structure. A rupture of the bladder would result in deflation, making the cabin inoperable, even by suited astronauts, because of the collapse of the tent around the controls. One potential solution to this problem is to replace the rayon cord winding with a thin layer of aluminum or other such metal, thus lending some rigidity to the structure without losing the weight and thermal advantages of the flexible tent design. This question probably merits further investigation, but for the purposes of the current study it was decided that a flexible cabin presented too many risks.

In order to avoid the problems inherent in the completely flexible cabin, a concept was devised which used a very small hard cabin, designed to satisfy only the driving function. An expandable rear compartment which is flexible would be used for all other functions. Figures 13.5-4 and 13.5-5 illustrate this concept in two of its roles.

A third configuration might be a traversing mode in which the inflatable structure is partially extended allowing part, or all, of the payload to be sheltered. This would essentially be a transporting-type mission, perhaps between a base and the tug. It should be noted that an inflated cabin, with suitable anchoring might be used for personnel transport between the tug and base.

Finally, a construction configuration might be imagined. The inflatable cabin is removed and other equipment such as a crane is mounted on the platform. The following points might be made in defense of this unit.

1. A hard portion of the unit must be available for hatches and the mounting of manipulators. Delicate electronics must be sheltered and hung on firm portions of the structure.

2. The addition of the flexible rear section provides for all of the other functions which are needed.

3. The partially-flexible removable cabin represents a considerable saving in weight over the totally hard cabin as represented by MOLAB. A crude estimate of this saving yields a figure of 500 lb. A more precise answer will have to await more detailed design.

4. As has been pointed out, the time required to perform an assembly of the space tent, assuming no cabin on the vehicle, will, in all likelihood, be approximately 1 to 2 hours. The proposed

configuration would require a matter of a few minutes for assembly. Further, it could be totally assembled while the men were in shirtsleeves. The tent must be assembled by fully-suited astronauts.

5. Unlike the MOLAB, the current concept provides for reasonable crew comfort in that a larger cabin may be used while retaining a saving in weight.

6. Obviously, the fact that the inflatable structure is removable implies that different-sized inflatable structures may be used. For instance, if the missions called for large numbers of EVA's but little sleeping or covered transport, the cabin could be used for sleeping and a small inflatable airlock be used in lieu of the larger structure.

One problem which is persistent is the weight distribution. Almost any small cabin design will, by its nature, be front-heavy. The payload may solve this problem. Another possibility is to place the fuel, power supply, and powering motors towards the rear of the vehicle. Another problem is the oxygen consumed in pressurizing the tent. One possible means of keeping this problem to a minimum would be to recompress the gas used in the air-lock utilization and in the sleeping quarters configuration. Trade-offs would have to be made between oxygen loss and compressor weight. Obviously, such considerations are beyond the scope of this study.

Other objections to the flexible living quarters include the rather marginal saving in weight realized by such a scheme, potential difficulty in fabrication of such a hybrid structure, and the rather inconvenient location of the airlock door, which would have required the men to climb over payload in some instances. It is the

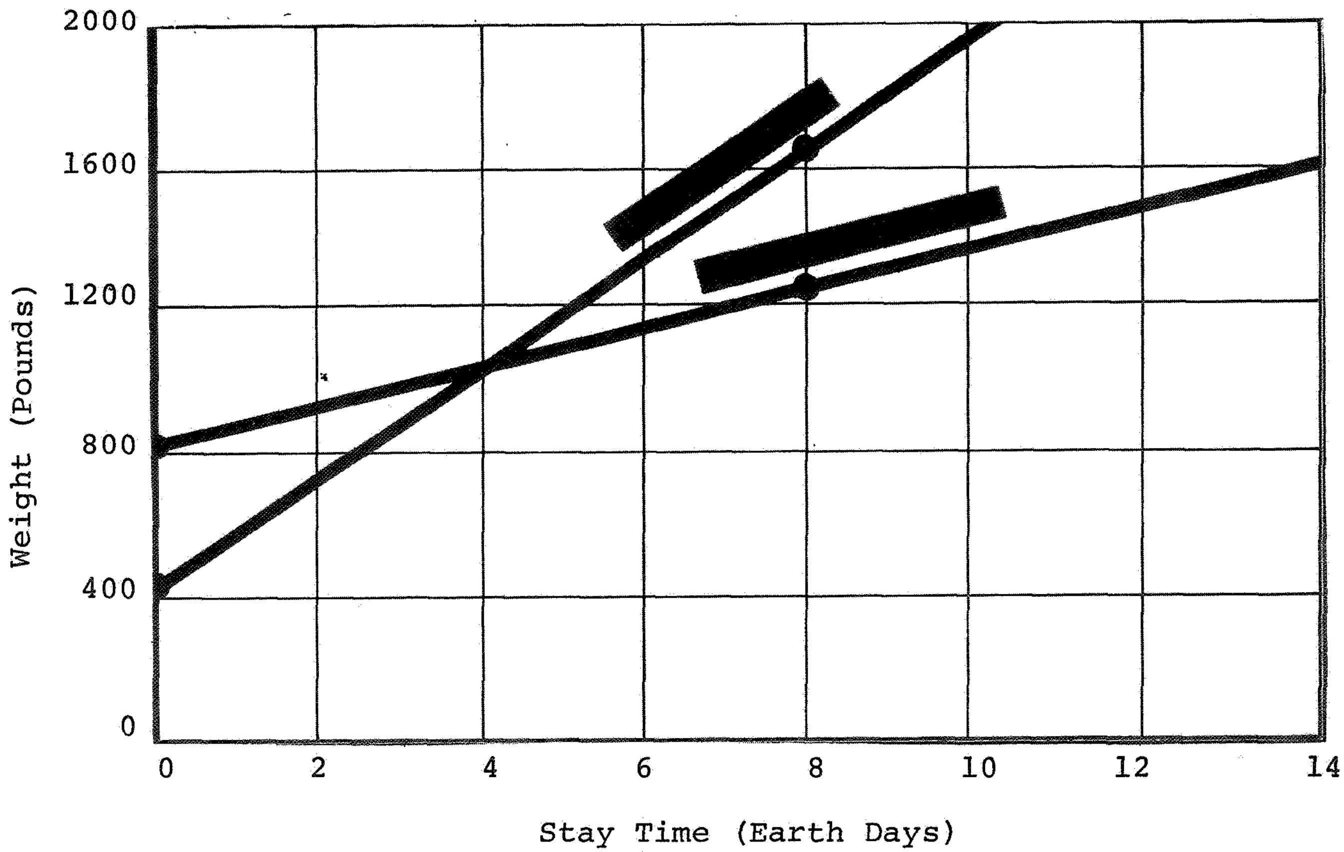


FIGURE 13.5-3 LUNAR TENT WEIGHT STUDIES[1]

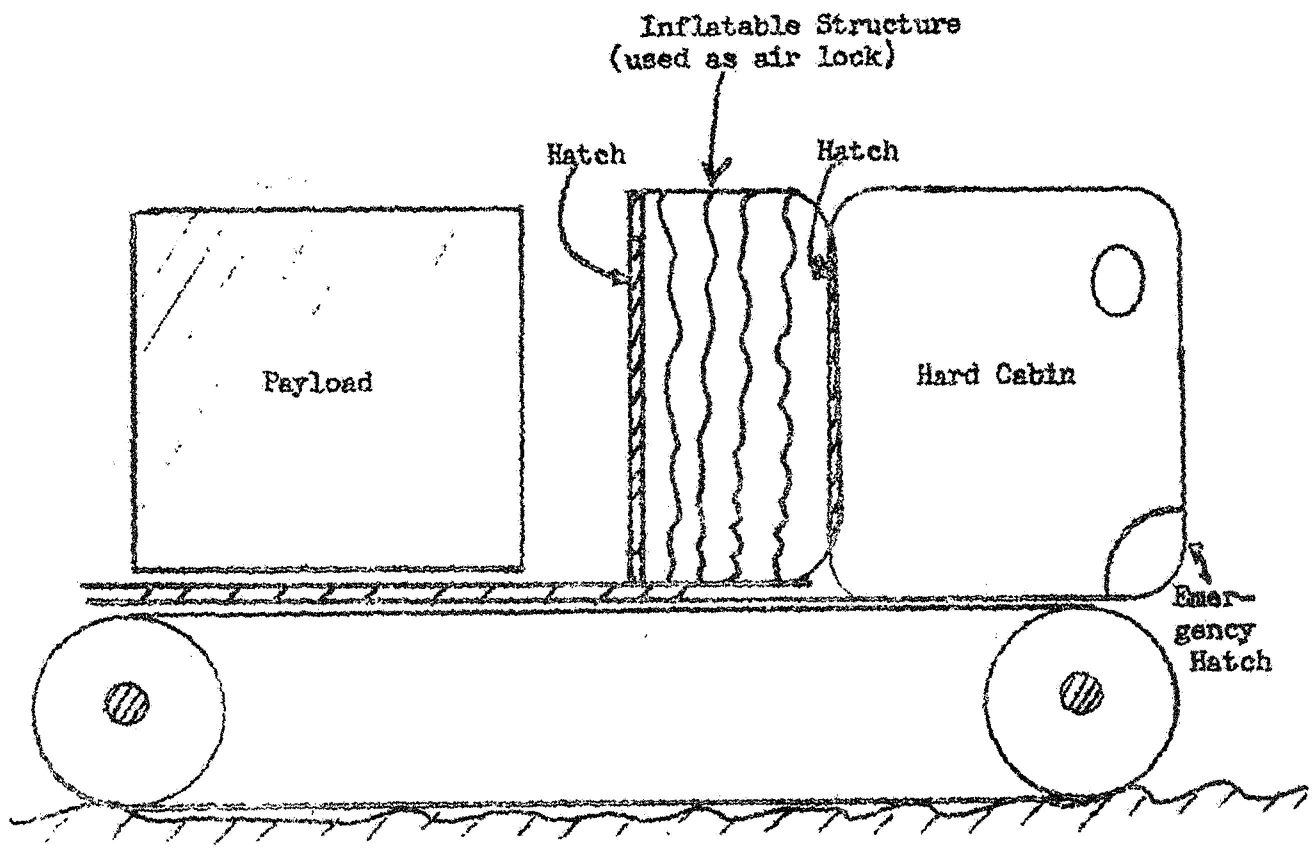


FIGURE 13.5-4   TENT CABIN CONCEPT IN DRIVING CONFIGURATION

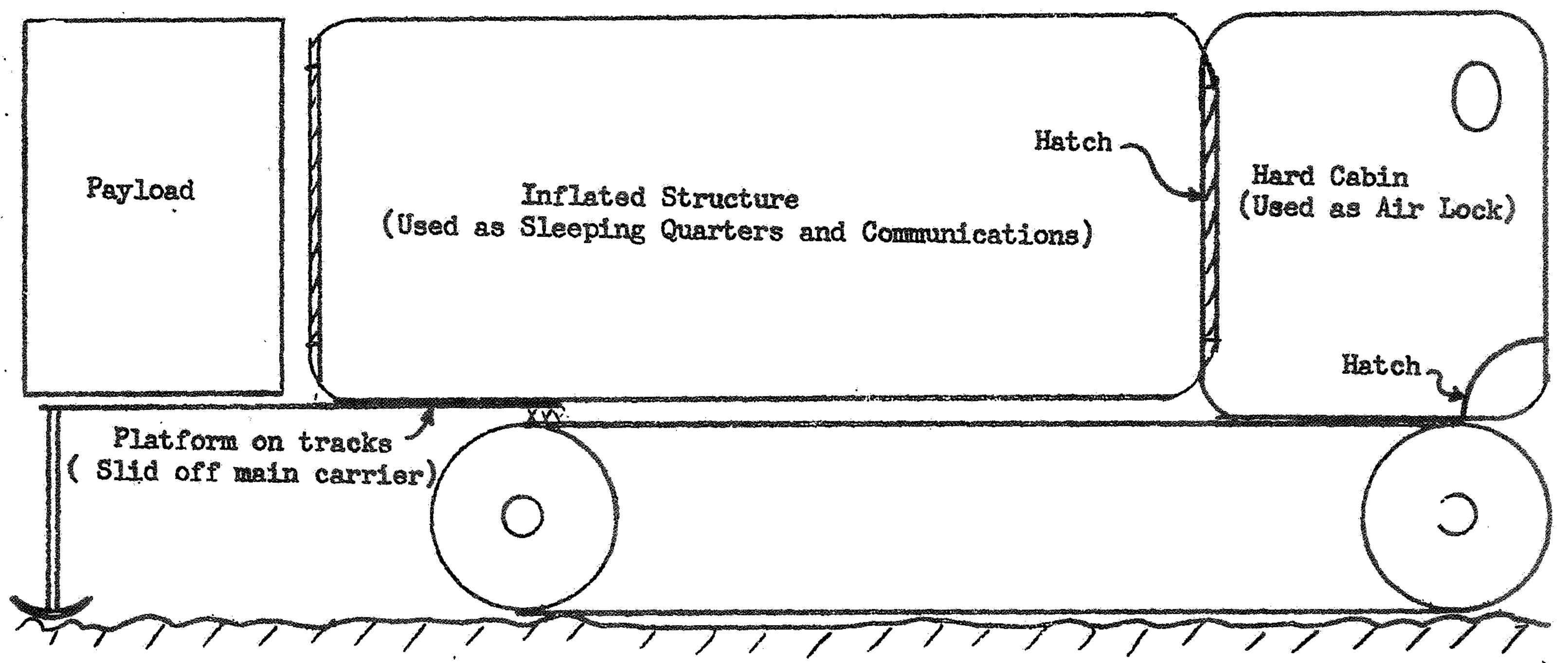


FIGURE 13.5-5 TENT CABIN CONCEPT IN TEMPORARY STATION CONFIGURATION

recommendation of the team that such a concept might well prove to be a fruitful one if some of its basic difficulties can be overcome.

In moving to a larger rigid cabin, in which the men could sleep and perform all other normal functions, some thought was given to the possibility of using the flexible-airlock concept. However, weight savings were marginal and the awkward placement of the airlock door remained a problem. All other avenues being exhausted, the group selected a completely rigid cabin as its candidate. It should be emphasized, however, that this is an area where substantial weight savings might be realized through an intelligent utilization of the flexible-tent concept.

The unique features of the final cabin design include fully reclining seats on which the men will sleep, bubble type windows which offer roughly a 210° viewing field, and an airlock with a door on the side and a retractable ladder upon which an astronaut may reach the lunar surface. Figure 13.5-6 gives a plan view of the general cabin layout, which indicates the features which are described in detail in subsequent sub-sections.

13.5.2 Cabin Windows

Several concepts were considered for the placement and shape of cabin windows. These include the type found in the MOLAB Cabin[2] which are recessed into the cabin sufficiently to virtually eliminate heat absorption problems except during the early portions of lunar day when the cabin is driving directly into the sun. This

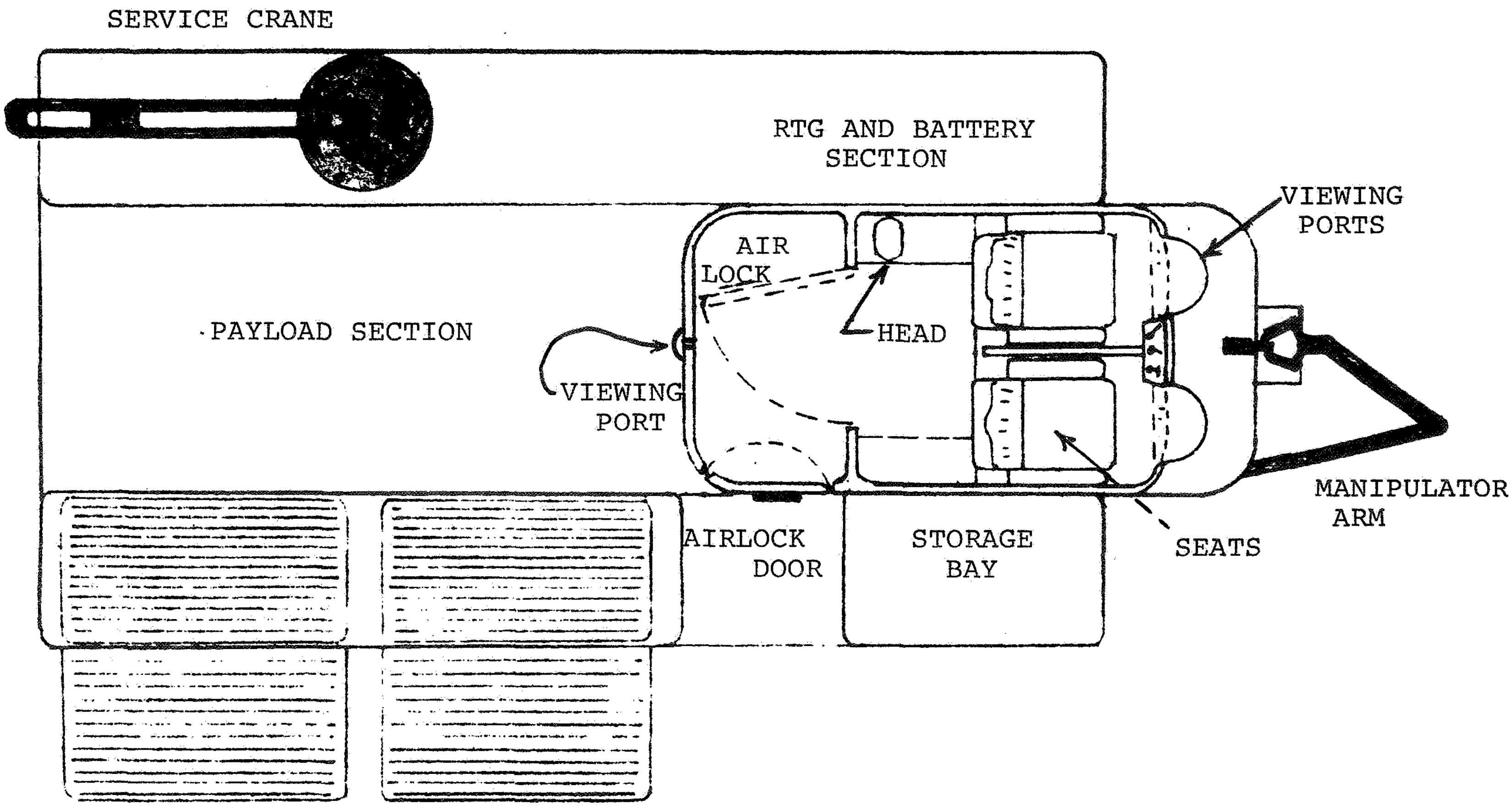


FIGURE 13.5-6 PLAN VIEW OF FINAL CABIN CONCEPT

window has the severe disadvantage that it limits forward vision to 40^{o} to the left and right and vertical plane vision to within 20^{o} of the horizontal. Its situation in the cabin is considered somewhat wasteful of cabin space. A bubble concept in the top of the cabin, in which the men would have a 360^{o} view, was discussed. Unfortunately this concept raised the center of gravity to an untenable height and caused severe heat transfer problems.

The final design is shown in Figure 13.5-7. It consists of two quarter spherical convex windows, placed side-by-side in the front of the cabin. These afford a greater-than 90^{o} view in both the horizontal and vertical directions; this eliminates the need for any viewing ports in the side of the cabin. Although the window arrangement is in a preliminary stage of design, a crossection of of the window can be rendered (Figure 13.5-8). There are two layers of polycarbonate windows with a vacuum space between the layers. A gold-plated visor is provided with a squeeze type edge. The basic idea is similar to that of the visor on the Apollo space suit. The edge of the visor will allow for the removal of dust from the windows. Moving from top to bottom, the visor may be held in any position, as is shown in the figure, by the pressure of the squeeze on the outer window. Exterior to the visor is a solid movable shield, made of the cabin material, which may be completely shut during periods when the use of the front window is not necessary. Finally, on the inside of the cabin are side-wise moving shields which shut from the side and meet at the center. Again, under severe lighting conditions these may be partially shut. Ob-

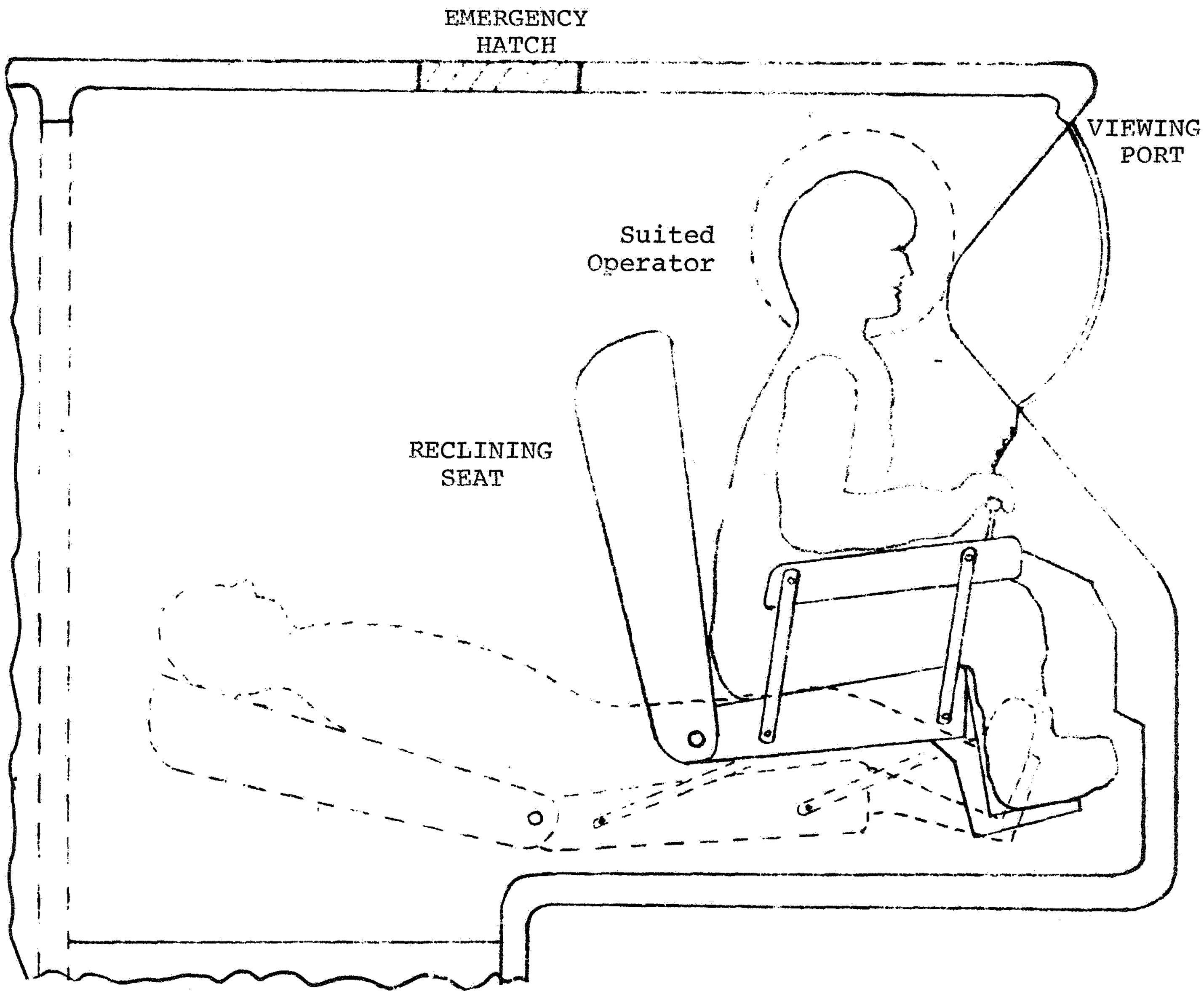


FIGURE 13.5-7 SECTIONAL DIAGRAM OF SOME FEATURES OF THE INTERIOR OF THE CABIN

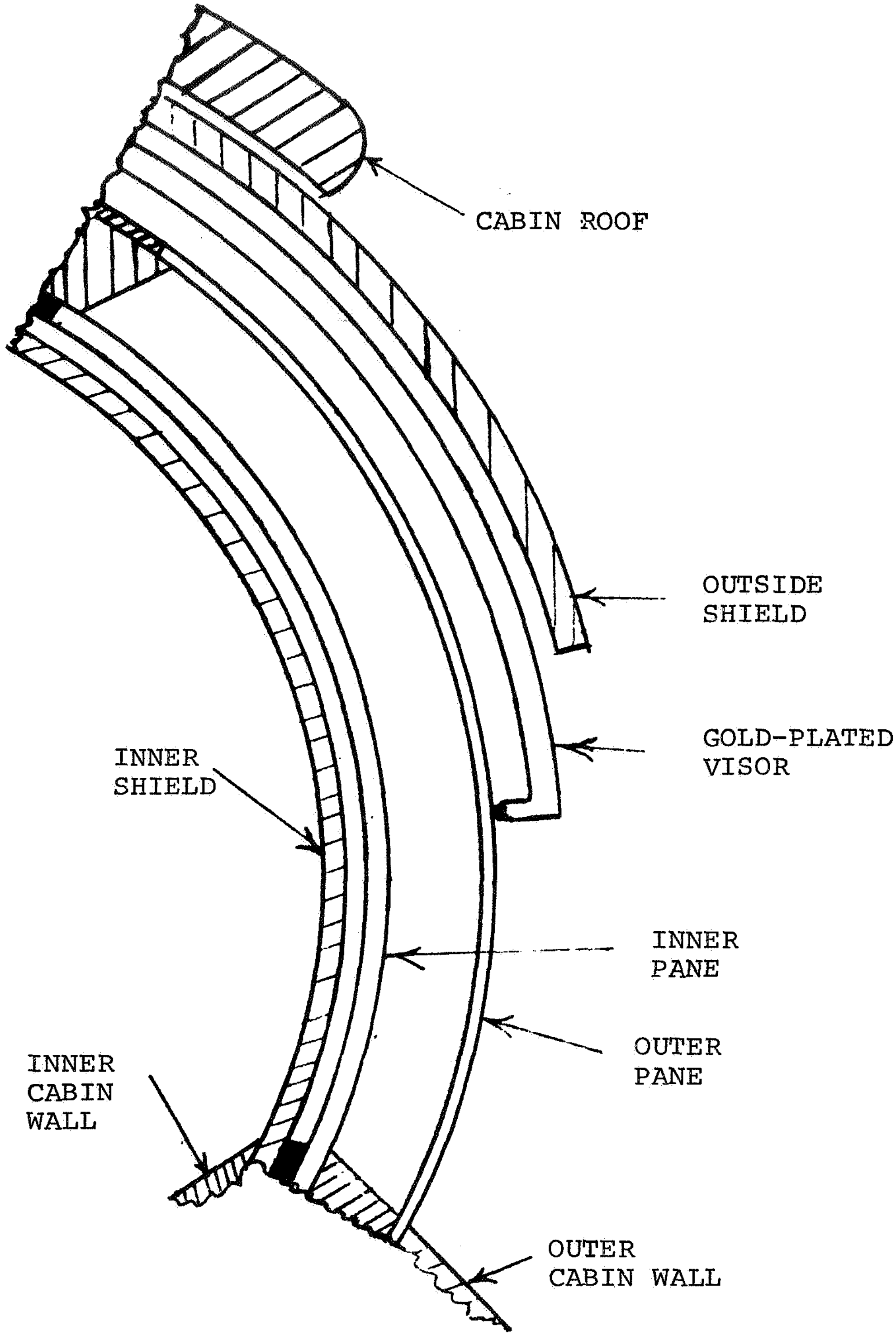


FIGURE 13.5-8 FRONT VIEWING PORT SECTIONAL DIAGRAM

viously during unmanned missions all the devices would be shut, thus providing thermal insulation which is about equivalent to the cabin wall. Since a final design could not be rendered in the time period allotted to this task, exact details of the implementation of this design such as precise accommodation of the shields to the cabin, operating mechanism of each portion and exact thermal analysis was not possible. There is, however, little doubt that the plan can be implemented.

One additional viewing port was provided in the airlock section. It is situated in the center of this structure at a height of approximately 5 feet. This port would find its primary use if the near television camera failed. It consists of a small lens-like structure which allows a wide-angled view of the rear of the cabin through a relatively small opening. It would be used in a manner similar to a rigidly-mounted telescope, with a "fish-eye" like lens. Again a shield on the outside of the port would allow for dusting and prevention of heat leak.

13.5.3 Seating

The dual-purpose seat was selected as being the most conservative of space in the cabin. Figure 13.5-7 illustrates its use. More care will be needed in the design of this seat than the conventional one, but the economy of space and weight makes this expenditure desirable. In the upright position a fully-suited operator may be accommodated in comfort. In the reclined position a good bed is available. Its basically flat configuration could allow for sleeping in several positions. Obviously, intermediate positions are possible.

13.5.4 Cabin Sturcture

The basic structure of the cabin consists of an outside wall made of a light metal such as aluminum with a coating, such as ZnO-based paint, which has favorable thermal properties. Inner layers include a thickness of superinsulation with a metal inner wall which serves as the cabin seal. Typical of such wall structure is the MOLAB[2] wall configuration.

An emergency hatch is located in the roof of the cabin. While it is slightly inconvenient, it does offer acceptable access in an emergency. A man could stand on the reclined seats to exit the vehicle. Again a typical design is anticipated.

The doors on the main cabin and airlock both open outward from the structure, as is shown in the cabin plan view (Figure 13.5-7). A ladder on the side of the track support structure allows the man to reach the ground.

13.5.5 Hygiene Facilities

The waste management facilities are contained in a compartment behind one of the seats. Urine will be collected as will all other waste water and be stored in a waste-water tank. This will provide a contingency supply of water for the water boiler in the coolant loop.

Solid waste will be contained in bags along with appropriate disinfectants. Probably they will be stowed for later disposal.

A source of chilled water, as well as limited quantities of hot water, will also be provided from the water-management system. These will be to a large extent used in food preparation, rather than personal hygiene. The latter activity will be minimized on 36-hour missions.

In summary it should be emphasized that the concentration of this program's activities upon conceptual design prevented a more detailed approach to the cabin. In its defense, it can be stated that the preliminary concept appears solid enough to merit proceeding with the design.

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CHAPTER 14

POWER

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P. S. Shieh

14.1 Introduction

Power Requirements and Power Supply Selection

The power subsystem refers to the power supplies and distributions for the MULE. The major power users are ECS and LS, navigation, power management, telemetry, locomotion, and science experiments. Table 14.1-1 shows an estimate of the average and peak power required for the operation of various subsystems.

TABLE 14.1-1
POWER REQUIREMENTS FOR SUBSYSTEMS (KW)

	Manned		Unmanned	
	Avg.	Peak	Avg.	Peak
Astrionics (Telemetry & Navigation)	0.1	0.2	0.1	0.2
ECS	0.2	0.5	0.2	0.5
LS	0.5	0.5	0.0	0.0
Locomotion	3.1	3.1	1.6	2.5
Power Management	0.1	0.2	0.1	0.2
Science	0.2	0.5	0.2	0.5
TOTAL:	4.2	5.0	2.2	0.5

For mission energy requirements we consider the manned and unmanned missions separately. The total energy consumption for unmanned mission by the ECS, telemetry, navigation, power management, and science experiements for a 420 hour mission amounts to approximately 924 Kwh. For a 36 hour manned mission, the total energy required is about 112 Kwh. These estimates are for lunar-day missions, which are at least five times that required for lunar-night missions. The total energies were estimated from the following schedule:

Manned Mission:	Science - 8 hours Locomotion - 25 hours Others - 36 hours
Unmanned Mission:	All subsystems are continuously operating at average power.

Having established the peak power and total energy requirements we now turn to the study of various types of power supplies and their characteristics. We shall choose a suitable power subsystem from a selection of (a) batteries, (b) fuel cells, (c) nuclear reactors, (d) RTG's, (e) solar cells, or (f) their combinations. A parametric check list is established (Table 14.1-2) to show the selection criteria for items (a) through (e) that are capable of supplying the required peak power and energy.

Based on the "check-out" data presented in this table, we conclude that the power subsystem shall be a combination of fuel cells, RTG's and batteries so that weight, interface, and life time problems of the subsystem can be overcome.

In order to select suitable "components" to form the power plant, we utilize the results listed in Sections 14.2, 14.3, and 14.4. Both SNAP-27 and SNAP-29 are in the same class of power-weight relations. They furnish approximately 600 W of power at a mass penalty of about 200 kg. For comparatively larger power output in a short duration we have to use batteries. The sealed silver-zinc batteries are chosen because of their high specific energy density. The power subsystem that satisfies the

TABLE 14.1-2
PARAMETRIC CHECK LIST OF POWER SUPPLIES*

	Batteries	Fuels Cells	Nuclear Reactors	RTG	Solar Cells
Safety	S	S	U	Q	S
Reliability	S	S	Q	S	Q
Mass	Q	Q	U	S	S
Volume	S	Q	U	S	Q
Subsystem Interface	S	S	U	Q	Q
Radiator Area	S	Q	Q	S	Q
Life Time	Q	S	S	S	Q
Reusability	S	S	S	S	Q
Environmental Effect	S	Q	Q	S	Q
Cost	S	S	Q	S	S
Maintainability	S	S	S	S	Q

*The symbols S, Q, and U indicate that the considered power supply is a suitable, questionable, and unsuitable mode for mission use, respectively.

power and energy requirements will then be made up of two SNAP-27 RTGs, four silver-zinc batteries, and an A-C, H_2-O_2 fuel cell with four modules.

14.2 RADIOISOTOPE THERMOELECTRIC GENERATOR

Radioisotope Thermoelectric Generator (RTG) is a power supply unit. The electric energy output of an RTG is derived from heat produced when alpha, beta, or gamma particles are slowed down or stopped. The major components of an RTG are (a) fuel elements, (b) thermoelectric convertor, (c) shield, and (d) container serving as radiator (Figure 14.2-1). The fuel elements are radioisotopic materials fabricated into capsules that emit energetic particles. Part of the kinetic energy of each particle is converted into heat which, in turn, is converted into the electric energy output of the RTG by the thermoelectric convertor. Normally, a shield is furnished to reduce the radiation leakage to a safe level. The entire system is then enclosed in a shell serving both as the containment vessel and the radiator for ejecting the waste heat. Extensive research in the development of this type of power supply is carried out in the Systems for Nuclear Auxiliary Power (SNAP) programs and others. In SNAP program, the RTG is usually designated by an odd number such as SNAP-3 or SNAP-29. Table 14.2-1 lists some of the SNAP RTGs with their characteristics. It is interesting to note that the specific power (watts/kg) of the RTG's increases in the recent SNAP models to close to 3 watts/kg. This limit

TABLE 14.1-3
POWER SUBSYSTEM DESCRIPTION*

Peak Power (kw)	5.0	3.9
Total electric energy (kwh)	112	924
No. of SNAP-27	2	2
Unit wt. (lbs)	40	40
Unit Vol. (ft^3)	2	2
Unit power (watts)	60	60
No. of Silver-zinc batteries	4	4
Specific Energy (w-hr/lb)	200	200
Volume energy density (kwh/ft^3)	5.0	5.0
Unit wt.(lbs)	12.5	12.5
Unit Vol. (ft^3)	0.5	0.5
No. of fuel cells	1 (4 modules)	1 (3 modules)
Module Unit Power (kw)	1.25	1.25
Specific Power (watt/lb)	34	34
Volume power density (kw/ft^3)	2.2	2.2

*See Table 14.5-1 for detailed weight and volume estimates.

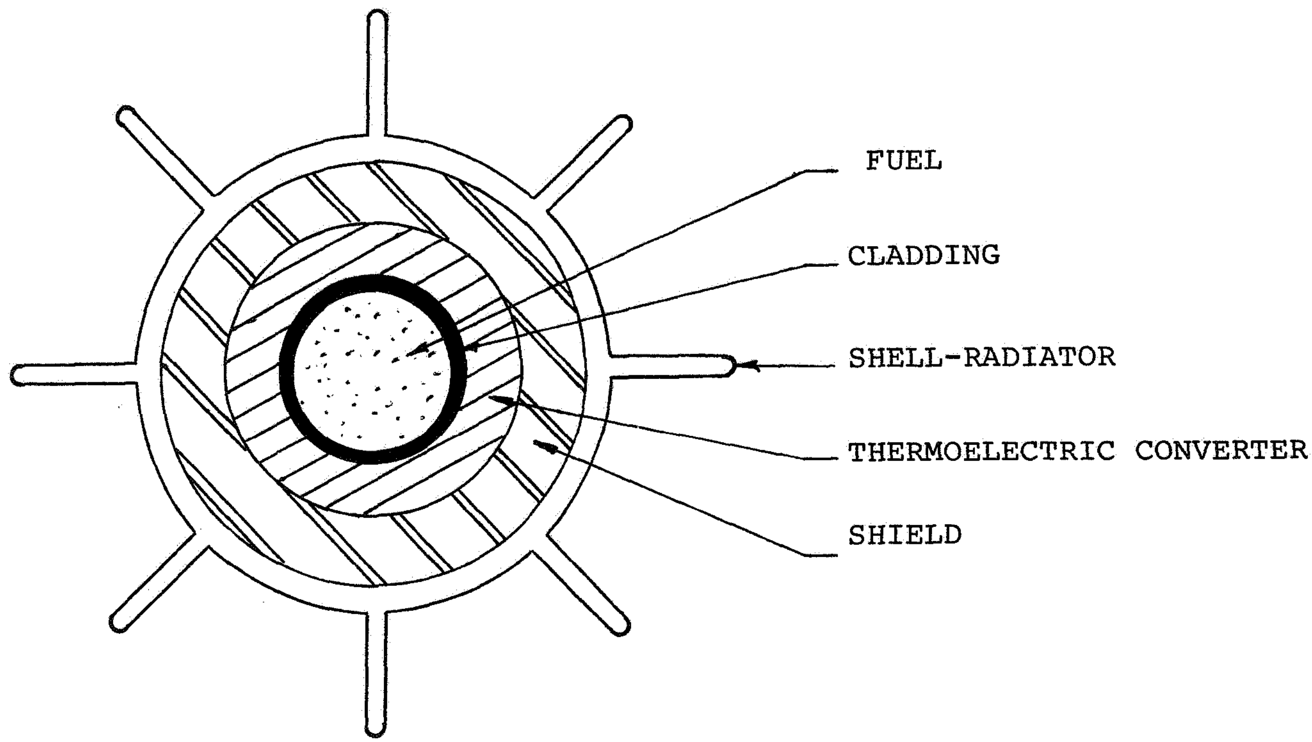

FIGURE 14.2-1 CROSS SECTION OF AN RTG

TABLE 14.2-1 SNAP RTG SYSTEM*

Designation	Designer	Output (Watt)	Lifetime (Yr)	Mass (kg)	Specific Power (w/kg)	Notes
SNAP-1	Martin	500	1/6	----	----	These models have never been comple-ly test ed and built.
SNAP-1A	Martin	125	1	91.0	1.4	
SNAP-3A	Martin	2.7	5	2.37	1.2	
SNAP-3A	Martin	25	5	12.3	2.0	
SNAP-3B+	Martin	2.4	---	2.3	1.0	
SNAP-7A	Martin	10	10	850	0.012	
SNAP-7B	Martin	60	10	2100	0.028	
SNAP-7C	Martin	10	10	850	0.012	
SNAP-7D	Martin	60	10	2100	0.028	
SNAP-7E	Martin	6.5	5	1000	0.0065	

SNAP-3B1 data
*This table is incomplete because some of the data are still classified information.

SNAP-9A	Martin	25	5	12.3	2.0
SNAP-11	Martin	25	1/3	13.6	1.8
SNAP-13	Martin	12.5	---	1.8	7.0
SNAP-15A	------	0.001	10	----	---
SNAP-17A	Martin	30	4	13.6	2.2
SNAP-17B	G.E.	30	4	13.6	2.2
SNAP-19	Martin	30	5	13.6	2.2
SNAP-21	------	10	5	----	---
SNAP-27	G.E.	60	1	18.2	3.3
SNAP-29	Martin	500	1/4	18.2	2.7
IMP-Generator	Martin	25	3	9.6	2.5
COMSAT		30	---	11.4	2.6

of the state of the art of the development of RTG is also the basic criterion used by us to estimate the power subsystem mass.

In general an RTG has the advantages:

a. Comparatively low total weight
b. Comparatively lesser radiator area required
c. Comparatively smaller volume required
d. Without moving parts
e. No fuel storage
f. Not affected by environment
g. Not directional (vs. solar arrays)
h. Highly reliable
i. Comparable costs
j. Long life time for operation

The last item is the most attractive feature of this device since, unlike a battery or a fuel cell, it does not require charging or fueling during a long mission.

Since an RTG is fueled by radioactive materials, radiation safety is a problem that must be analyzed carefully during the design stages of such a package. This disadvantage prevented the use of RTG's in the early days of space programs. However, due to advanced materials research and design, it is feasible to utilize the RTG as a power source in space applications now. In fact, for the later model of the SNAP RTGs, the escape radiation level of a few MREM/HR at a distance on one foot from the source is well under the safety limit.

14.3 BATTERIES

Batteries are usually classified as primary or secondary. In the latter case, repeated charging and discharging is possible so that a secondary battery can be reused. For space applications, batteries must possess the following characteristics:

a. Large number of cycles (CHG/DISCHG) to failure

b. Long-wet-stand time

c. Light weight

d. Small volume

e. High-energy-density

f. High current capability

g. Reliability

i. Wide temperature stability

These properties embrace the basic concepts of both a primary and a secondary battery. Consequently, a battery system for space programs such as MULE must be such that it is not only rechargeable as a secondary battery but also capable of furnishing high current for a short duration as a primary battery.

There are four battery systems under development. They are (a) nickel-cadmium, (b) silver-cadmium, (c) silver-zinc, and (d) lead-acid. Some of their characteristics are presented in Table 14.3-1. One major disadvantage associated with a battery system is its susceptibility to the operating temperature. For example, the percent of original capacity of silver-zinc battery decreases to about 50% when the battery is operating at -50^{O} as shown by the Figure 14.3-1.

TABLE 14.3-1 PROPERTIES OF BATTERY SYSTEMS

Battery Type	Amp hr lb.	Wh/lb	Wh/in^3	No. of cycles to failure*
a.	70	100	1	9000
b.	100	130	2	7500
c.	110	200	3	220
d.	---	10	-	----

*at 75°F and 50% depth of discharge.

Since the energy density of silver-zinc battery is the largest among the four systems considered, we have chosen this system for the MULE, even though there are problems such as short life time, small number of cycles to failure, and capacity loss at operating temperatures outside the nominal temperature range of 0°F - 150°F. We believe these problems can be overcome under intensive research. For example, a passive method of preserving the operating capacity of the battery is simply using an insulator to maintain a constant operating temperature.

14.4 FUEL CELLS

One of the methods of converting chemical energy into electric energy is to use a fuel cell. In general there are three classes of fuel cells: namely direct, indirect, and regenerative. For the MULE design we selected fuel cells using H_2 and O_2 as the reactants are introduced into the reaction chamber (reactor) which may be kept at an operating temperature by thermal control. The chemical energy is directly converted into electric energy which is extracted from the cell electrodes. For steady state operation, water is produced at the same rate as the rate of consumption of reactants (H_2 as fuel and O_2 as oxident). Figure 14.4-1 shows a schematic diagram of a fuel cell system.

A fuel cell supplies electric energy, within the designed capacity, proportional to the reactant consumption as shown by Figure 14.4-2.

The water production of a fuel cell and the necessary storage of O_2 as oxident are the factors that make the fuel cell - LS integration possible.

In an attempt to select a suitable fuel cell system for MULE we examine the six models listed in Table 14.4-1.

In order to attain maximum power with the least amount of cell system weight and volume we choose the Allis-Chalmers model for the MULE power subsystem. This model has an almost constant voltage output. Therefore, much of the power quality and quantity control difficulties are reduced.

14.5 SYSTEM VOLUME AND WEIGHT SUMMARY

The reactant volumes and weights need now to be determined and the entire power system volume and weight summarized. The fuel

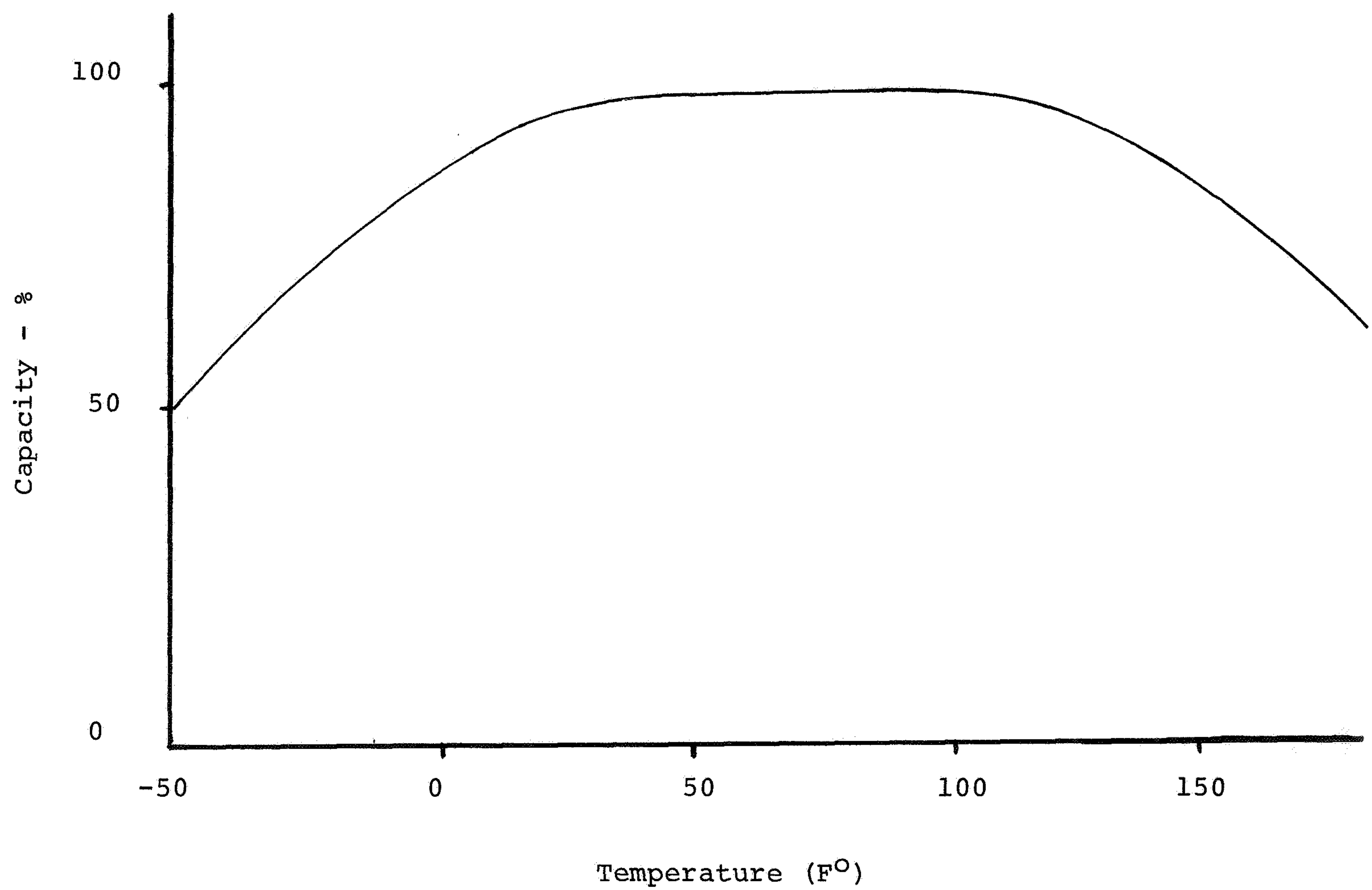

Figure 14.3-1 TEMPERATURE EFFECT ON A SILVER-ZINC BATTERY

TABLE 14.4-1 FUEL CELL CHARACTERISTICS ((H_2-O_2) SYSTEMS)

Designer	Peak Power (kw)	Power/wt (w/lb)	Power/vol (kw/ft^3)	Tested Operation Time (Days)	Operating Tem. (°C)
G.E.	1	15	0.7	42	60
Union Carbide	1	50	2	210	60
Allis-Chalmers	5 (4 modules)	34	2.2	92	90
Westing-house	1	10	1	84	250
Bacon-Pratt & Whitney	5	10	1	84	250
Justi-Varta	1	11	---	---	60

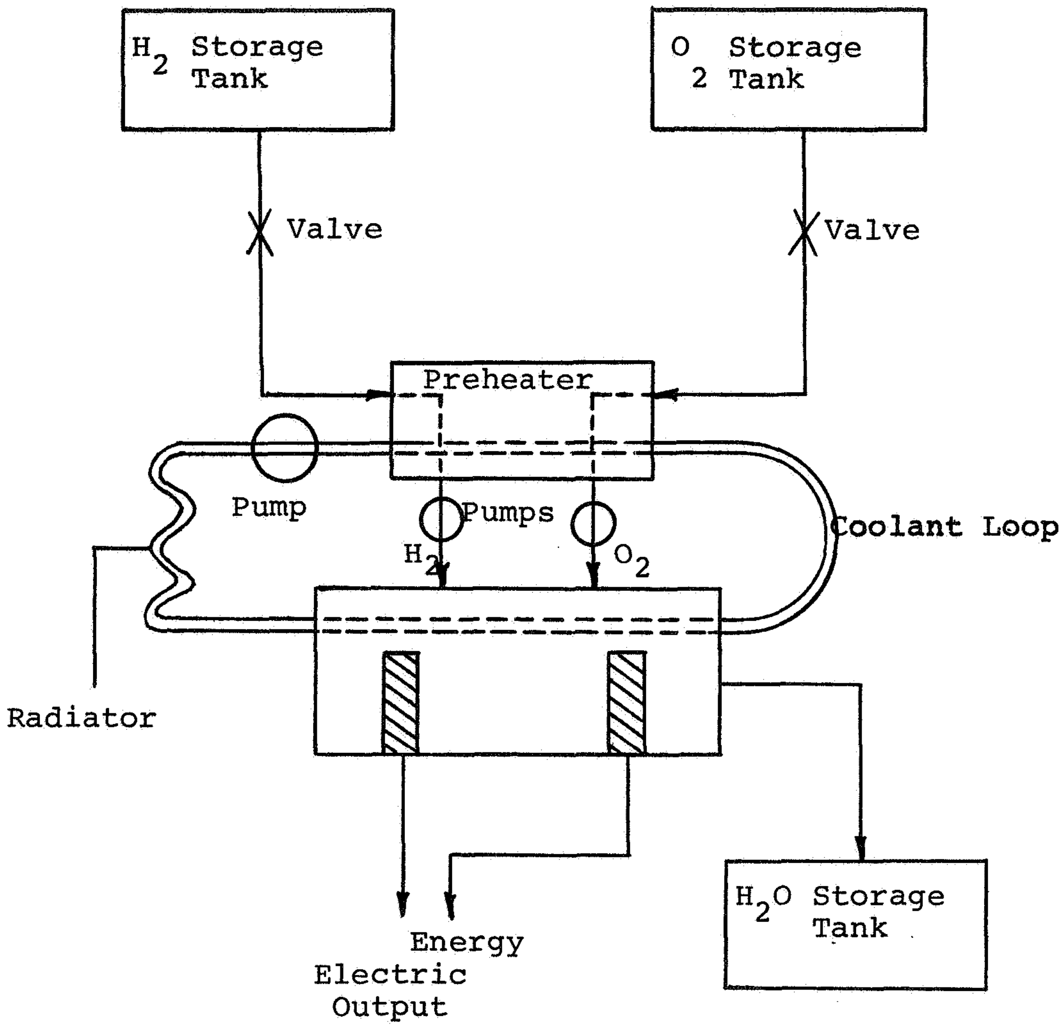

FIGURE 14.4-1 SCHEMATIC DIAGRAM OF A FUEL CELL

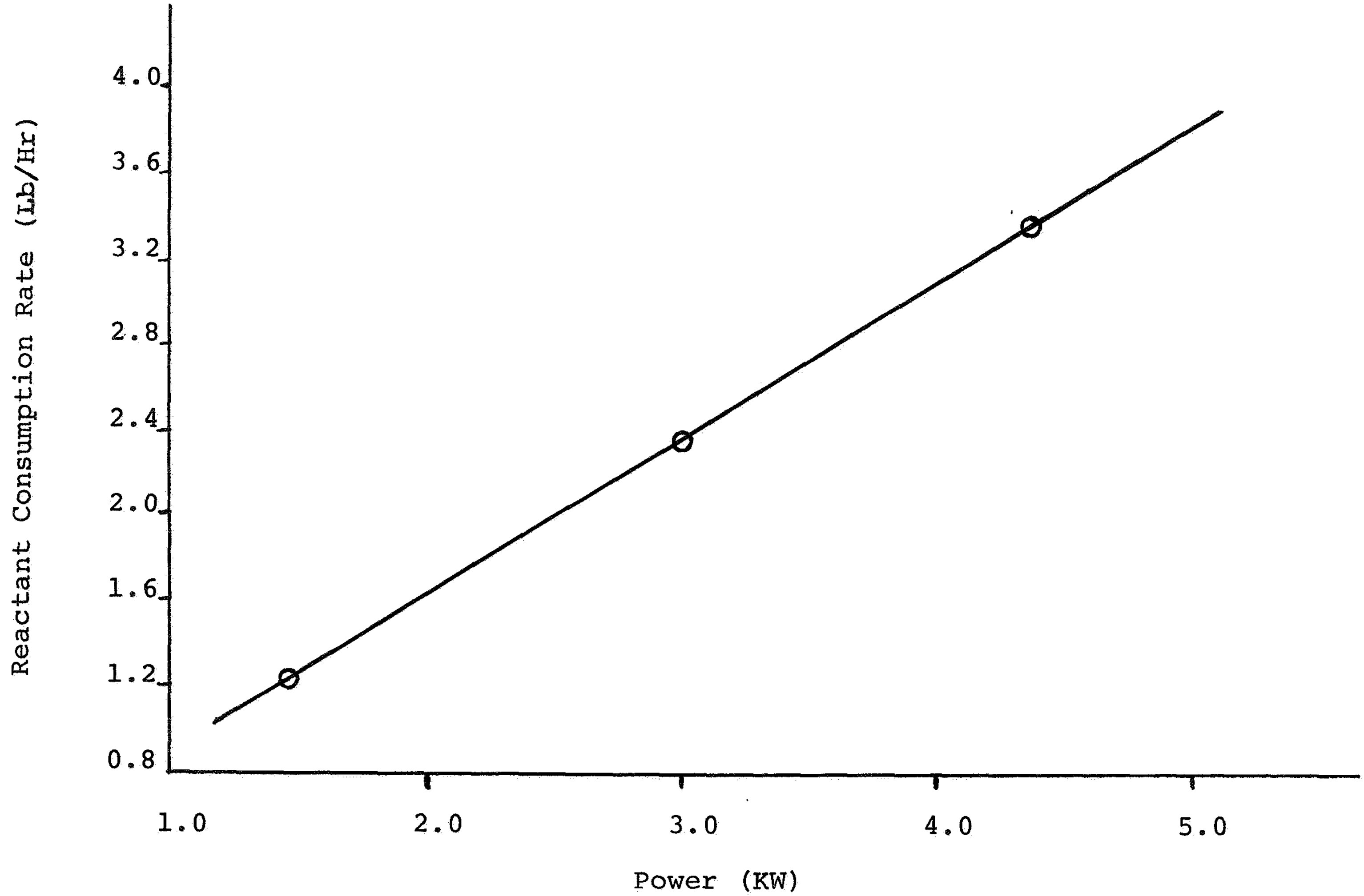

FIGURE 14.4-2 REACTANT CONSUMPTION OF A FUEL CELL

cell reactants, O_2 and H_2, will be considered first. For the unmanned mission, the total energy required was previously determined to be 924 KWH. The specific reactant consumption rate, S, is approximately .8 LB/KWH. Thus,

$$W_R = SE\ TOT \qquad (14.5\text{-}1)$$

$$= 1.25 \times .8 \times 924 = 924 \text{ lb. of reactant}$$

The 1.25 is the required contingency factor. The oxygen and hydrogen combine in the mass ratio of 8:1 to form water. Therefore,

$$W_H = 103 \text{ lb. (weight of } H_2)$$

$$W_O = 822 \text{ lb. (weight of } O_2)$$

The stored volumes of these reactants are:

$$V_H = 121 \text{ ft.}^3$$

$$V_O = 37 \text{ ft.}^3$$

$$\text{TOTAL} = 158 \text{ ft.}^3 \text{ of reactants}$$

A quick look at the energy requirements for the manned mission shows that much less fuel is required. Thus, the design volume requirements are dictated by the unmanned mission. Using the Molab report[6] as a reference, the tank weights were determined. The complete summary of power system volume and weight data is shown in Table 14.5-1.

14.6 POWER MANAGEMENT

When electric energy is extracted from power supply units for applications, one has to adjust its quality and quantity according to the demands of the user. We shall regard this quality

TABLE 14.5-1 POWER SYSTEM VOLUME AND WEIGHT SUMMARY

UNIT	UNMANNED MULE VOL (ft^3)	UNMANNED MULE WT. (lb)	MANNED MULE VOL (ft^3)	MANNED MULE WT. (lb)
Fuel Cell Modules	1.7	110	2.3	147
Radiator	6	85	6	85
H_2 Fuel	121	103	14.6	12.4
O_2 Fuel	37	822	4.5	99.6
H_2 Tank	-	575	-	143
O_2 Tank	-	313	-	77
Cooling System	1	14	1	14
SUB-TOTAL	167	2022	28	578
RTG	4	80	4	80
BATTERIES	2	50	2	50
POWER MGT.	8	160	8	160
SUB-TOTAL	14	290	14	290
TOTAL	181 ft^3	2312 lb.	42 ft^3	868 lb
Dry Weight		1387 lb		757 lb

and quantity power control as a power management task. Furthermore, we also include the following in this task:

Power subsystem configuration

Power distribution and transmission

Power failure insurance

Based on the power subsystem description we construct the power subsystem block diagram as shown by Figure 14.6-1.

The major functions of the different types of power supply units are summarized in Table 14.6-1.

TABLE 14.6-1 POWER SUBSYSTEM COMPONENT FUNCTIONS

COMPONENT		
RTGs	1.	Charge batteries continuously at a slow rate
	2.	Supply "raw" power to be regulated by PCU
Batteries	1.	Furnish starting power for locomotion
	2.	Supply high-current for short durations
Power Control Unit	1.	Adjust power quality
	2.	Effect DC-DC conversion
	3.	Furnish switching capability
	4.	Insure single point failure ability
Bus		Distribute power
Cable	1.	Furnish locomotion power
Fuel Cell	2.	Produce water
	3.	Supply power for LS, ECS, and Science

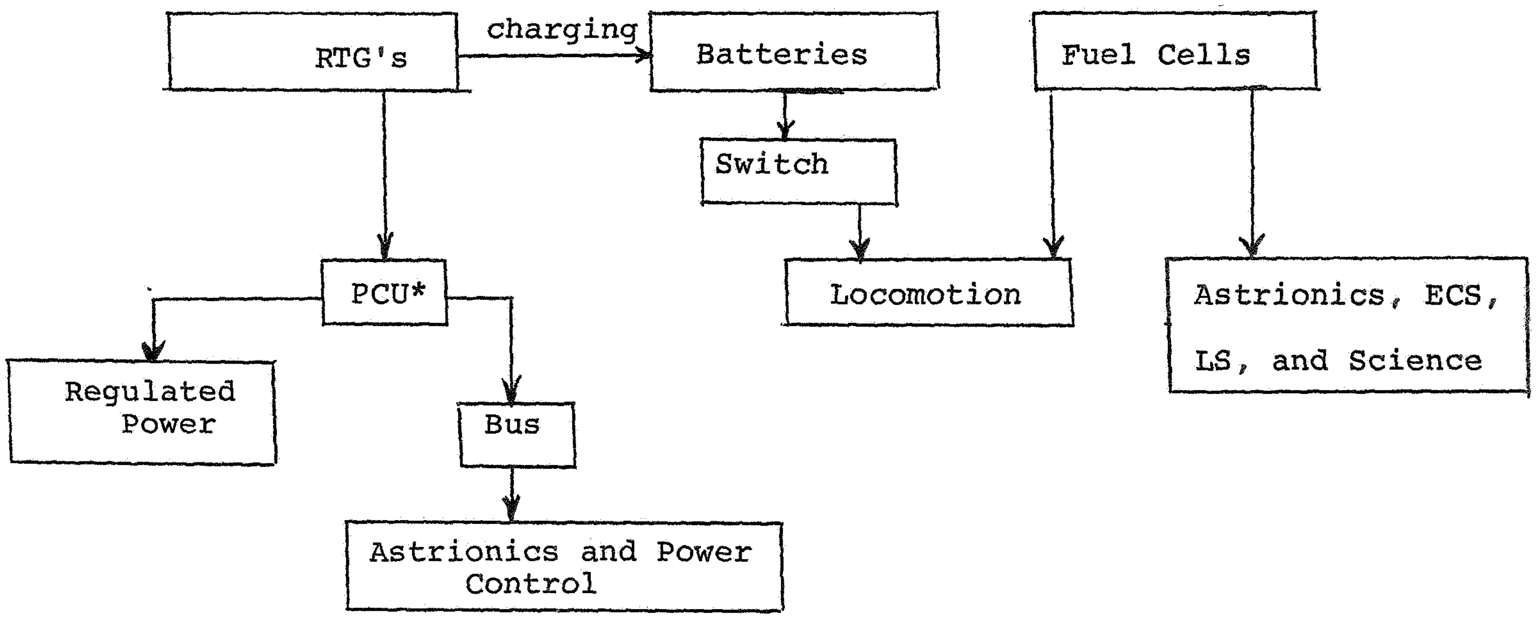

FIGURE 14.6-1. POWER SUBSYSTEM BLOCK DIAGRAM

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CHAPTER 15

ASTRIONICS

G. Lawrence

The astrionics subsystems include nearly all the electronic systems of the MULE except for the power supplies. The requirement of maintaining communications between all stations results in a sophisticated on-board communications system plus the satellite systems described in Chapter 19. Accurate navigation is essential in support of science packages, and for aiding the operator in finding his destination. Visual displays, such as television and obstacle sensors, are desirable for manned operation and essential for remote driving of the MULE. Remote driving over long distances (1000-1500 km) is one of the major challenges in the MULE design, requiring elaborate computer logic both on the MULE and at the remote station. Following is a description of these subsystems.

15.1 Communications Subsystem

L. C. Ludeman

The MULE communications subsystem must transfer voice, telemetry, video, data and commands to support operation of the MULE in both the manned and unmanned modes of operation. In order to describe the communication subsystem, the communication points and links must first be specified. The possible earth-moon configuration in the 1980-1990 period has already been shown in Figure 5.2-1 and the main terminals are: the earth to earth orbit shuttle, Earth

Orbit Space Station (EOSS), Nuclear Shuttle, Lunar Orbit Space Station (LOSS), Space Tug, Lunar Base (LB), Extra Vehicular Communication System (EVCS), Lunar Satellite System and the Manned-Unmanned Lunar Explorer (MULE).

To simplify the specification of the communication network the following guidelines were established:

(a) Direct communications from the MULE to the Space Shuttle, EOSS, and Nuclear Shuttle were judged unnecessary. Indirect communication could be easily obtained through earth relay or LOSS relay if necessary.

(b) Access to full time coverage for both near and far side lunar remote control of the MULE can be obtained through a Satellite System composed of a "halo" Satellite around libration point L_2 and a "humming bird" Satellite around libration point L_1. An explanation and comparison of various lunar satellites along with a justification of the selected system appears in Chapter 19.

(c) Since in the experimental stages of the MULE, the satellite system and the Lunar Base might not be functional, the vehicle must be able to be controlled from earth. In the middle of the decade, LOSS, when fully established, will assume prime control for the unmanned operation while the Lunar Base or surface lander will assume control for the manned operation. In this way the links from earth, LOSS and LB will carry almost identical information but since the links are used one at a time, the MULE communications subsystem require but one hardware section.

From the above guidelines, main links along with the informational transfer were established and are shown in Figure 15.1-1. The TV link to the MULE should be thought of as optional while the other information is necessary for satisfactory completion of the missions.

The selection of the frequency bands for the links depends upon existing MSFN capabilities, projected bandwidths, state of art, etc. The following guidelines for frequency band selection were established:

(a) Because of the suggestions from the surface transportation description Document 1, it is recommended that the MULE's communication system be a singular carrier S-band to earth. Because of this it is recommended that all non-lunar surface terminals be S-Band also.

(b) As the CSM/LM, earth communication systems are functional, the use of them as a basic system provides commonality and reduces the design and development costs. These systems, however, need to be modified to accommodate MULE system requirements.

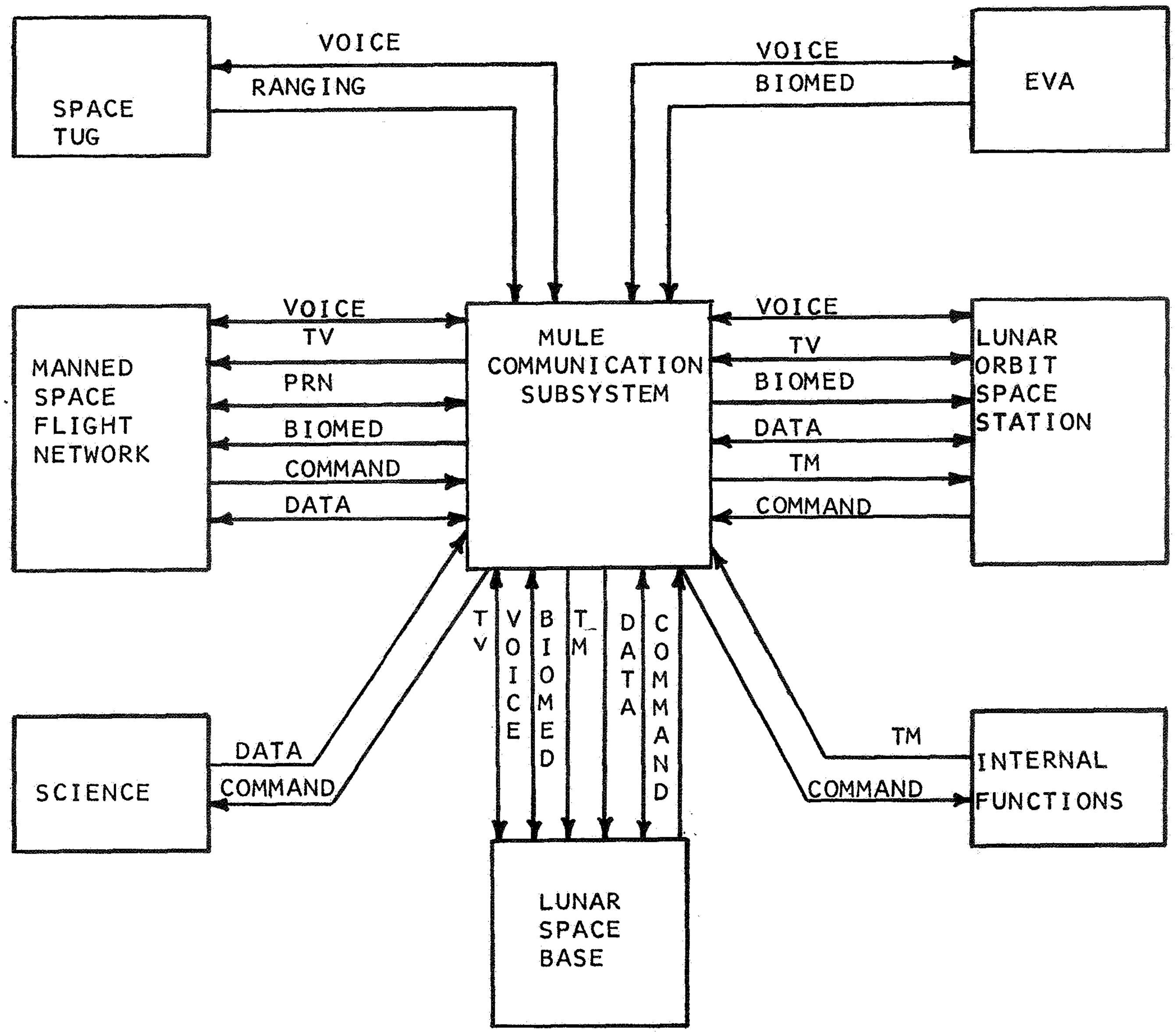
(c) Because of the existing VHF equipment in LM and EVCS it is felt that surface activities for voice and low data rate science will use VHF equipment.

Therefore, the communication subsystem is broken into two main sections, the S-Band section and the VHF section. A more detailed discussion of the two sections is now presented.

15.1.1 S-Band Section

The S-Band section is used for all non-surface activities. The

FIGURE 15.1-1 MULE COMMUNICATION SUBSYSTEM INTERFACE



Command Service Module (CSM) communication system, a unified S-Band system, serves as a basis for the design of the MULE communications subsystem. The Apollo system (2) up link (from Earth to CSM) contains on a S-Band carrier the following: Pseudo Random Noise (PRN) range code; up-data for updating information for the onboard computer or real-time or stored program commands; and Voice with bandwidth of 300-2300hz. The Apollo down link (spacecraft to ground) consists of the recovered and retransmitted range code; pulse code modulated telemetry having either 51.2 kilobit or 1.6 kilobit information transmission rate; Voice with 300-2300 hz bandwidth; Biomedical data from the astronauts; television, not commerical quality, at 320 lines per frame and a 500 khz bandwidth; Recorded PCM telemetry at either a 1.6 Kbs or 51.2 kbs rate; Recorded voice; Emergency voice and emergency key. The Apollo spacecraft S-Band transponder consists of three basic parts, the receiver and two transmitter exciters, one PM and one FM. TV is always transmitted Frequency Modulated (FM) and the range code always Phase Modulated (PM); Real time voice and telemetry is either FM or PM but together, while recorded voice and telemetry is either FM or PM but together and not on the same channel as the real time voice.

Communication requirements of the MULE that are different from the present Apollo S-Band system are:

(a) At least one down link commercial grade television channel for remote operation and if possible two channels are necessary.

(b) Increased data transfer due to science interface.

(c) A video for operation with the boomcrane while loading and unloading.

(d) The system must be compatible with relay satellites that will be necessary in the mid 1980's.

Each of these requirements alters the present Apollo System. The first requirement for one commercial TV link for remote control can be satisfied by an increase in the receiving and transmitting band width to about 5 mhz. This will also dictate increasing the power of the FM transmission and modification of the present ground receiving system. The addition of another commercial grade TV for panning presents a problem. A possible solution is the addition of another FM exciter to the present apollo CSM communication system to be time shared with the PM transmitter range code. This is possible as the range code is necessary for a small fraction of the total mission time.

The second requirement can be satisfied by time sharing the ranging and the additional experimental data. This required only a change at the ground station receiver and the proper switching in the premodulation processer. By making the video boom monitor a low frame rate, low resolution device, the above ranging channel could also be used to satisfy the third requirement.

Use of relay satellites probably would require power modification of the S-band communications system. However, to make any reccommendation would require a complete link analysis which was not possible within the time frame set for this research.

Preliminary results by Philco-Ford[3] show that the Apollo system gives a positive link margin for down link communication but a negative link margin for up link communication via satellite systems. A trade off should be made between satellite design and MULE communication power but will not be undertaken here.

The recommended communication subsystem for the MULE appears in Figure 15.1-2 and is a modified Apollo system. The VHF section not mentioned here is discussed in section 15.1.2.

The S-band communication subsystem consists of a directional and omnidirectional S-band antenna. The directional high gain S-band antenna must be capable of maintaining continuous pointing to the selected MSFN station or controller as the vehicle traverses over slopes of up to 45°. This requirement forces the antenna to be mobil in the total upper hemisphere. Even at this, communications directly to earth in the regions around the lunar poles will be poor, if not unobtainable. This antenna must track the station it is communicating with for high data rate transmission and video. This tracking might be done by special tracking antennas or by using the omnidirectional information. In the manned operation when high data rate transmission is not necessary, the low gain omni antenna can be used without tracking. Boeing[4] has shown that to avoid interference between antennas the omni antenna should be placed 0.85 meter above the directional S-band antenna. This placement is shown in Figure 11.4-10.

The modulation processor in Figure 15.1-2 is a signal processor which functions to accomplish the signal modulation and signal mixing of the information to be transmitted from the spacecraft and demodulation of the up-link voice and up data. The inputs are recorded voice and data, biomed, science data, video, etc.

15.1.2 VHF Section

The prime purpose of the VHF will be the completion of the controller station/astronaut link and the astronaut/astronaut link. The system must be designed such that astronaut/MULE communication has a maximum range of 5km. as specified in the mission requirements. This range will fix the power requirements and antenna sizes necessary for satisfactory VHF communications. The MULE to EVCS will be voice transmission and the EVCS-MULE will be voice and biomed.

In the Apollo communications system the voice of one astronaut was relayed through the EVCS system of the other astronaut and as this was an added on design it is felt that a redesign could eliminate the relay and its resulting problems and restrictions. The redesign requires only a rearrangement of transmitters and receivers already designed for the VHF Apollo system, and at the same time results in a savings in electronics. This rearrangement is shown in Figure 15.1-3. The MULE/EVCS-1, EVCS-2 links are shown where R1, T1, R2, T2, R3 & T3 represent different frequency amplitude modulations (AM) receivers and transmitters. A 4km range is suggested rather than the 5km proposed since 5km would require a large increase in EVCS power according to Bendix[5]. For the MULE VHF section a power of 1 watt with a 6ft. whip antenna is recommended to give the 4km range. This also requires

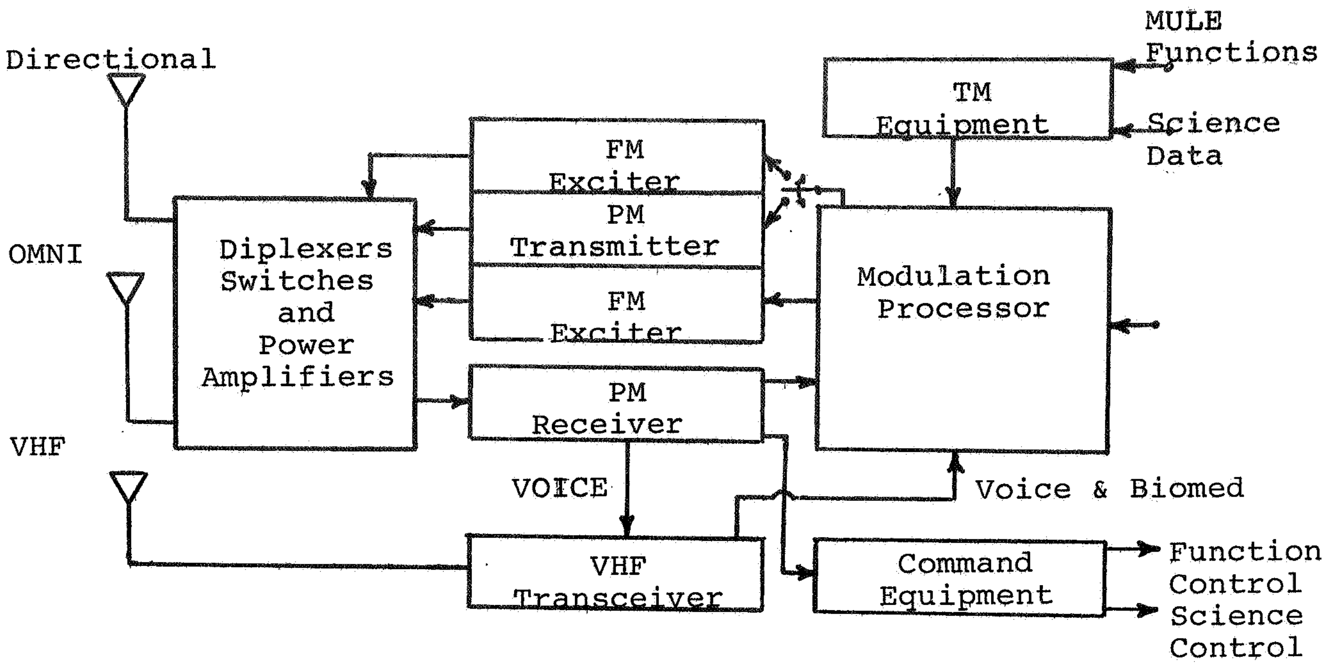


FIGURE 15.1-2 BASIC MULE COMMUNICATION SUBSYSTEM

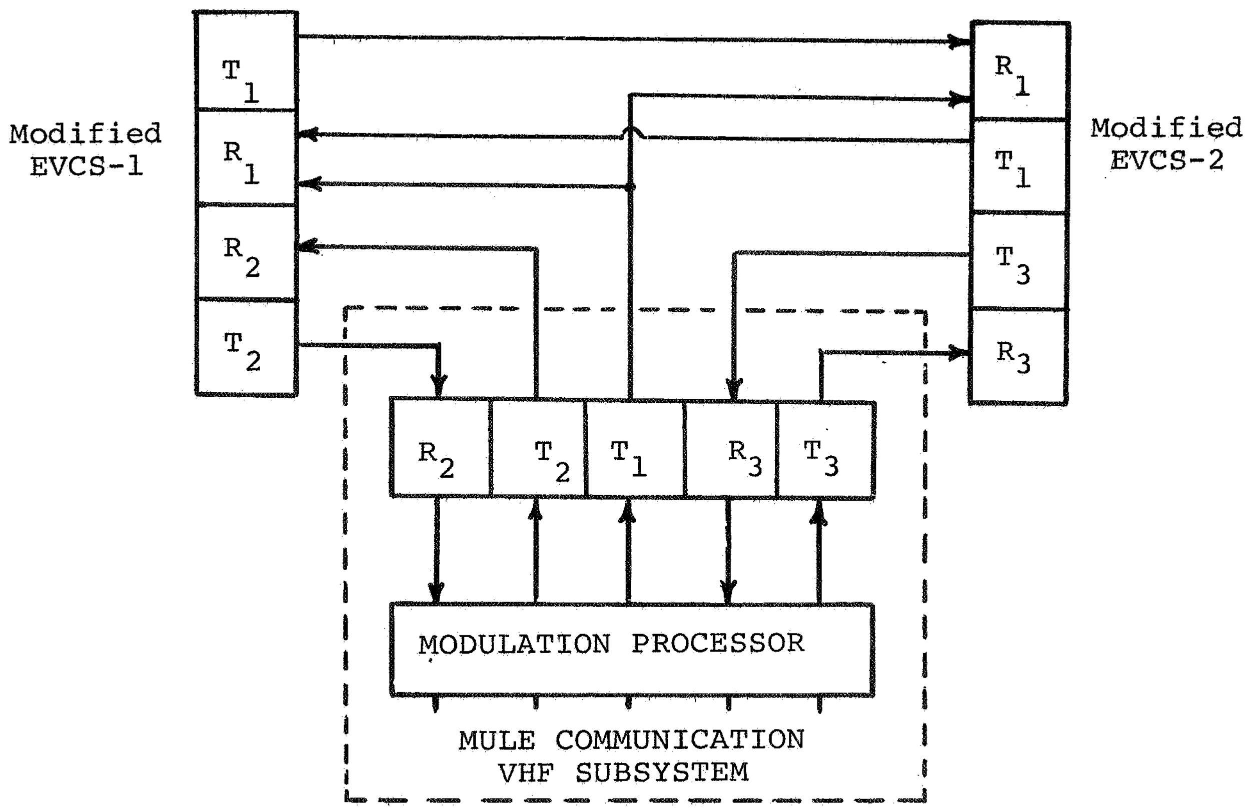


FIGURE 15.1-3 VHF TRANSMITTER-RECEIVER ARRANGEMENT

the EVCS system to increase the power output to 1 watt and have a 6 ft. whip antenna.

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15.2 Navigation

G. Lawrence

Navigation implies determining the location of the vehicle. For our system, navigation requirements are derived from two different functions:

1. Certain science experiments demand accurate location knowledge, on the order of $\pm$ 10 meters. (denote "precision navigation")
2. As the vehicle moves it is desirable to know position with respect to destination or origin, typically within capability for line-of-sight recognition of destination--about 500 m at eye level. (denote "routine navigation")

The latter is required continuously while the vehicle is moving long distances (greater than 1 km) and is within the capability of on-board sensors and processors. Precision navigation is required only occasionally, which is fortunate because almost all of the possible techniques are time consuming and require external aids.

15.2.1 Precision Navigation

Several methods have been investigated to determine position accurately. Each requires aids which cannot be built into the vehicle.

(a) Earth-based Triangulation from 3 earth stations requires transponders on the vehicle and at a fixed base, to receive and re-transmit coded signals. Relative position between the fixed base and the MULE could be determined within 40m[1]. On the order of 5 minutes is required to determine position. The required equipment, though very complex, is located on earth except for the small transponder packages. The coded signal uses a bandwidth of 5 mc. The methods are developed only for the near-side of the moon (line-of-sight); conceivably far side measurements could be accomplished with accurately placed satellites, but this has not been studied.

(b) Lunar based triangulation using ground wave propagation does not require line-of-sight, but there are uncertainties in ground wave propagation with respect to propagation time and attenuation. Again, the vehicle would carry transponders to retransmit a coded signal from fixed lunar stations. Accuracy on the order of 50 m is predicted.[1]

(c) Lunar based tracking using line-of-sight relays may be practical, if a permanent set of communication/tracking relays could be spaced at 20 to 30 km intervals on the lunar surface. For a typical base mission approximately 20 relays would cover the area within 80 km of the base; another 40 relays would cover a 1000 km path to the next base. Overall precision would depend on how accurately each relay station could be located; within 50 m is a predicted accuracy[1].

(d) Star fixes apparently provide accuracy only within a km or so.[2]

(e) Photographic mapping offers the most accurate means of position fixing. Current photographic maps from orbiting satellites permitted location of Surveyor within 10 m, and 1 m resolution is feasible by 1980.[2] Operating procedure would be to scan surrounding landmarks and compare with maps. If the traverse is well planned, the appropriate maps could be carried in the vehicle; a more complete library of maps would be stored in the LOSS, on earth, or at a lunar base. The vehicle position is likely known within 500 m from the routine navigation, so a precision fix could be obtained quickly from maps; however, lunar optics could conceivably make landmark recognition difficult.

At this time, highest precision and most flexibility in precision navigation would come from using photographic maps. As noted in Section 15.4, the obstacle avoidance data can aid in determining position accurately. Earth-based ranging provides a backup with little weight penalty.

15.2.2 Routine Navigation

Routine Navigation would use on-board sensors and processors to continuously update the vehicle location. A standard system uses an odometer on a track wheel to obtain speed and total distance traveled. A gyro is used to measure direction with respect to lunar north. Another gyro measures vehicle attitude with respect to horizontal. The navigation computer uses this data to determine range and bearing to the home base and/or destination. The accuracy depends on the time and distance between calibrations; to maintain accuracy within 500 m the gyros and odometer should be recalibrated approximately every 15 km or 1 hour, using the precision navigation method outlined above.

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15.3 Remote Control

Richard R. Pikul

A remote control system with man in a relatively safe control station appears desirable and feasible for future lunar and extraterrestrial exploration. Although the primary advantage of a remotely operated system is the elimination of man operating in a hazardous environment, another important consideration is the savings in weight and power achieved by eliminating the life support hardware required for manned operation. Still another advantage results when one considers that extended traverses will tire one or two onboard operators but remote personnel can work in shifts and do not need to meet the rigid physical requirements of astronauts to perform their tasks. If handled by several operators, even tedious missions can be performed without significant boredom resulting in lax vehicle control. Furthermore, if the remote station is earth, individuals competent in specialized areas can be called upon to aid in control when unusual conditions arise.

The primary disadvantage of remotely controlled systems is that man's unique sensory system cannot be duplicated over extended distance with possible time anomalies. Man's eyes are replaced by cameras, his ears by microphones, and at times his brain by a computer operating logically, but incapable of interpreting unique or novel data. Further distortion occurs in the display which provides the final link to man's own sensory system. Such sensory displays result in degraded perception which leads to less than optimal control in most cases.

Primary factors in a remote control system are visual sensors, visual displays, time delays and obstacle avoidance systems. It is assumed that the best remote system utilizes subsystems common to both the manned and unmanned modes, e.g. the automatic hazard avoidance subsystem aids in sensing and emergency maneuvers in either mode.

The primary feedback in remote vehicle control is provided by the visual sensors. In general, visual detection of hazards depends on object size, color contrast, object contrast against the lunar background and viewing angle. Contrast on the lunar surface is generated by shadows, albedo differences and surface slopes. Contrast caused by surface slopes, or photometric contrast, is characterized by high contrast for low sun angles and low contrast for high sun angles relative to the sensor. High contrast results in a surface being brighter than the horizontal. When the sensor viewing angle and the sun angle are at 0^{o} phase angle, or within 5^{o} of each other, reflectance is so high and shadow detection so minimal that features become indistinguishable over this 10^{o} area. This phenomena, known as washout, is illustrated by the Apollo AS11-6115 to 6118 frames which show the lunar feature known as the Cat's Paw disappearing from view as the lunar module viewing angle passes through the sun angle [4].

Shadows appear best at sun angles less than 30^{o}, with the sun in front of the sensor. However, with extremely low forward sun angles, glare causes poor visibility. Sun angles greater than 70^{o}

result in very little shadowing and hence poor crater definition. Additional viewing problems result when the sensor looks upslope, in which case craters on the flat portion of the slope are not distinguishable at moderate distances.

The combination of lens angle and sensor height determines the total field of view. If the depth of field is not greater than the product of velocity and time delay, visual obstacle avoidance is impossible as the operator can never see an obstacle in time to take diversionary action.

Although sensing may be adequate for a particular camera configuration, the display of sensed material to the operator provides the final link in the visual communication loop. Standard two-dimensional visual displays reduce depth perception ability, especially if the horizon or known landmarks are not visible to provide distance cues. Another consideration is the number of sensors to be displayed at one time. In the case of only one display, diversionary action could be complicated by a lack of visual information on the characteristics of the lunar surface outside the field of view of the single sensor.

Operation in the lunar night requires artifical illumination with problems similar to those mentioned previously if the illumination vector and viewing vector are within a 5° phase angle.

When an operator is required to control a vehicle remotely from a considerable distance, e.g. earth to moon, the transmission and information processing time becomes a major problem. He is no longer controlling the current movement of the vehicle based on current information. He is, instead, affecting the future path of the vehicle based on past information. For example, a lunar surface vehicle operated remotely from the earth must operate through a minimum one-way time delay of approximately 1.5 seconds. This results in the operator viewing information which is 1.5 seconds old and making control movements which will not affect the vehicle for another 1.5 seconds. To control the vehicle accurately the operator must predict where the vehicle will be 3.0 seconds (1.5 + 1.5) from the location depicted on the visual display.

Man's ability to control such a vehicle is dependent upon several factors including the speed of the vehicle, the time lag involved, the roughness of the surface to be negotiated, the a priori knowledge of the surface, and the type of display or displays available to the operator. Requiring a man to work with time delay while attempting to control a vehicle traveling at a moderate to high rate of speed (3-7 km/hr) over a rough surface is pushing him beyond the limits of reliable operation. Experimental work conducted at Stanford University [1,2,3] has resulted in an indication of operational limits for remotely controlled vehicle systems in a time lag situation. The basic results of this research coupled with other information inferred from this study are presented in Figure 15.3-1.

From this figure the following conclusions can be made. The feasible speed of a remotely controlled vehicle is highly dependent upon the distance the operator is from the vehicle (time delay). The

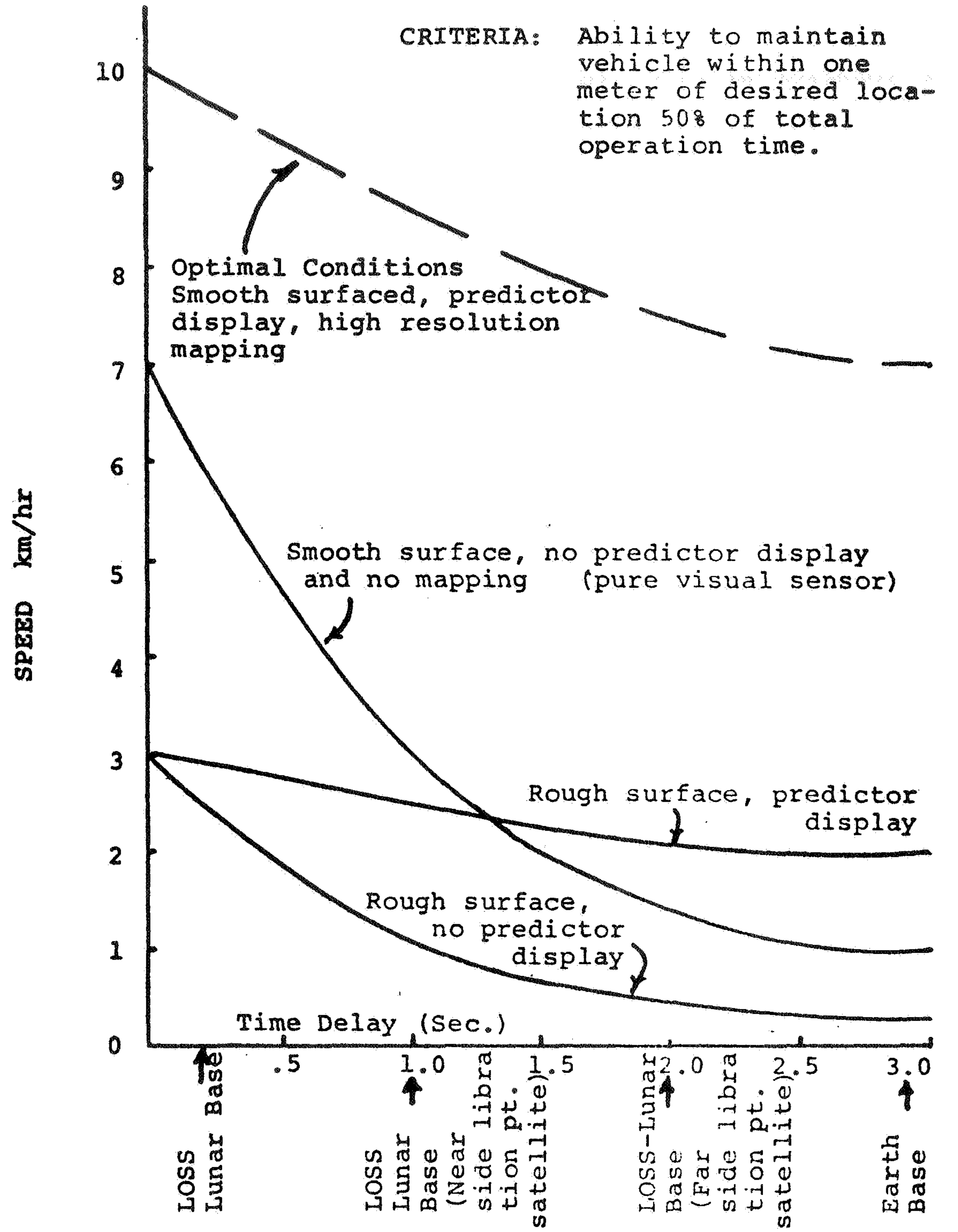

FIGURE 15.3-1 SPEED vs TIME DELAY

use of predictor display techniques which give the estimated vehicle location at time $t+\theta$ rather than time t relieve the man of making such predictions thereby resulting in his being able to control the vehicle in what appears to him to be a no-delay or real time situation. From the figure it is evident that the use of such displays has a positive effect on vehicle operating speed. The use of high resolution mapping will increase the accuracy of the predictor displays and hence will also increase performance.

Remote operation of a lunar vehicle from the earth even with high resolution mapping and a smooth surface for operation will result in a maximum speed of approximately 7 km/hr. The speed can be increased by placing the operator in a lunar orbiting space station (LOSS) or a lunar base. While this cuts down the basic time delay it results in certain areas of non-operation since line of sight communication is assumed. This line of sight problem also occurs when operating the vehicle from the earth and is one reason for the processing time delay. The line of sight problem can be solved with an increase in time delay by deploying libration point satellites or a system of lunar orbiting polar satellites. However, for the near lunar control stations the total time delay will still be less than that for earth based control. Further information on the satellite systems is available in Chapter 19 of this report. Even under optimal conditions the maximum practical speed for operating the MULE remotely is approximately 10 km/hr which is approximately the average speed expected when operated in the manned mode.

Based on the problems associated with remote vehicle control and items to be considered in the problem areas the following recommendations are made for the MULE system. Basically, a system using two visual sensors, preprogrammed traverses, obstacle avoidance subsystems, and predictor displays is recommended. Photographic maps of the lunar surface, compiled from pictures taken by orbiting satellites, are presently available with 10 meter resolution and one meter resolution is predicted for 1980. From these maps a traverse path will be selected which minimizes the fuel required for climbing, turning, and negotiating rough terrain. The path will be selected to avoid obstacles, allowing as much clearance as possible but as little as 1 meter clearance will be tolerable. The complete path (up to 1500km) along with 100 meters on each side will then be recorded on a film strip compiled from the pertinent satellite maps.

The maneuvers by the vehicle to follow the selected path will be specified, coded into approximately 16-bit words, and stored until the actual mission. Whether to store the complete traverse program (roughly 30k words for 1500km) in an on-board memory, or to transmit segments of the program from the remote station, provides a trade-off between communication links and on-board storage capacity. It is felt that more flexibility is provided by storing the complete program at the remote base and then transmitting segments (e.g., 500m sections) to a memory buffer in the vehicle at 15 minute intervals.

Even though the nominal path is carefully planned, there will be unexpected obstacles on the order of 1 meter size, which are not detectable from the photographic maps. Also deviation from the nominal path due to vehicle dynamics or sensor errors can result in collision with an obstacle. To prevent this situation from occurring two techniques are employed.

The traverse will be monitored by a remote operator using television displays with inherent visual problems mentioned previously. An on-board hazard avoidance system will also be used to automatically sense obstacles and provide preprogrammed avoidance maneuvers. For example, if a large crater is detected 15m ahead, slightly to the right of the vehicle path, the avoidance computer will command the vehicle to slow down, bear left until the object is passed, then return to the preprogrammed path. If a complicated sequence of hazards entraps the avoidance logic, or if the deviation from the programmed path becomes large, the vehicle will stop and wait for manual control. If a short range (2 meters) sensor detects a hazard, the avoidance computer commands an emergency stop and waits in a standby mode for operator control. For details on the avoidance subsystem see Section 15.4 of this report. Displays at the remote station will include the photographic map strip mentioned previously, plus two TV monitors presenting color displays of the vehicle path. A conceptual configuration for the displays is shown in Figure 15.3-2. The projection will display a 200 x 500 meter strip with hazards and nominal path emphasized. A display processor will generate a predictor symbol to represent vehicle position and heading pro-

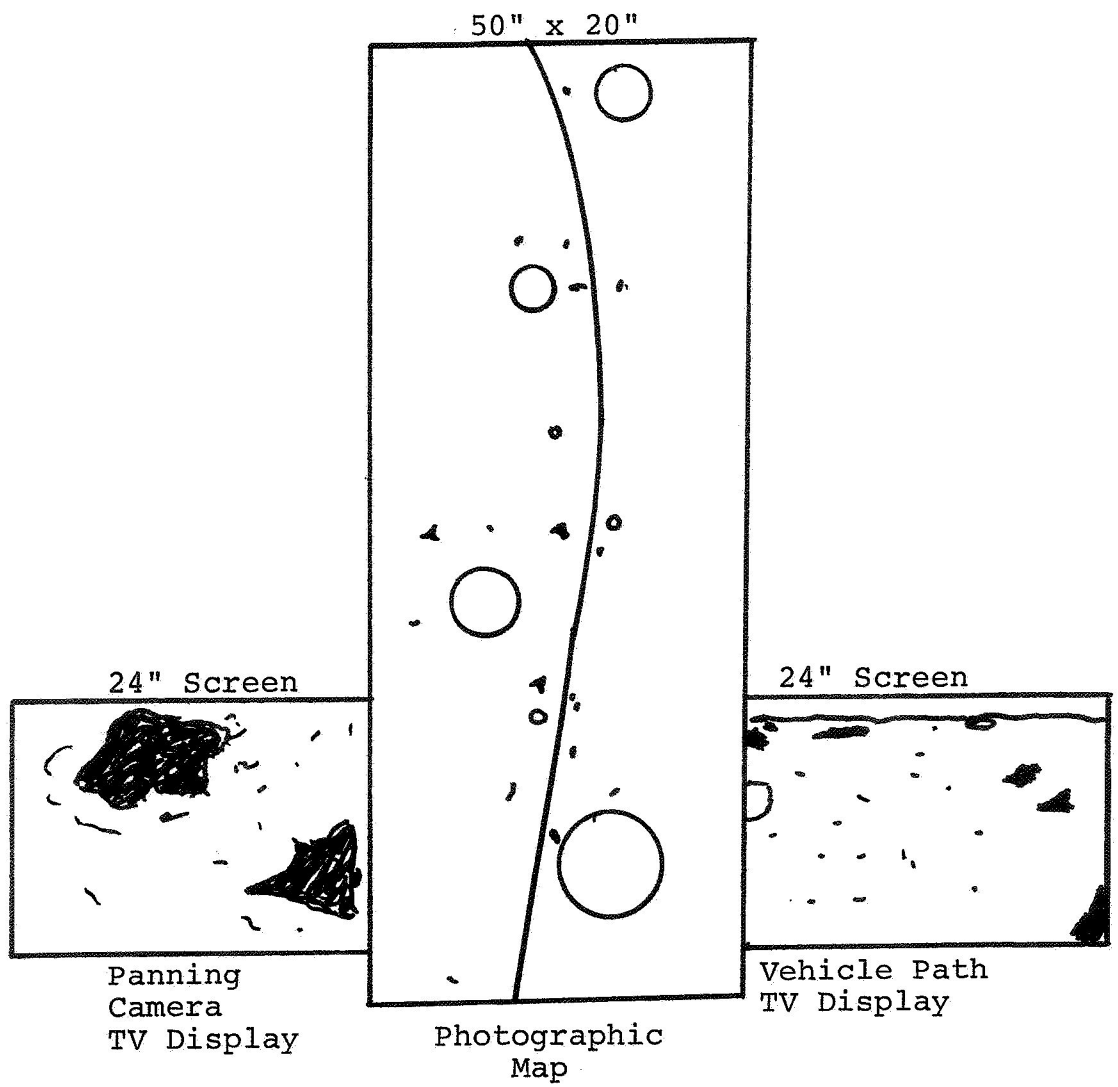

FIGURE 15.3-2 CONCEPTUAL REMOTE CONTROL DISPLAY

jected for the instant a remote command becomes effective at the vehicle (one-way communications delay). Prediction, based on vehicle sensor data and programmed maneuvers, is necessary because of time delays mentioned previously. This predictor display will enable the operator to manually control the vehicle to follow the preselected path as if there were no delays. For the physical presentation the map will be moved down as the vehicle moves forward and the predictor blip will be moved laterally for left and right motions. One TV monitor will show the path ahead of the vehicle, the display of the forward onboard videcon camera mounted to point in the direction of vehicle motion. The field of view extends from approximately 2 inches ahead of the vehicle to the horizon, and includes about 65^{o} horizontal azimuth.

The second TV monitor is operator controlled and has remote panning, tilt, and zoom capabilities. It will be used to augment the forward camera for obstacle avoidance maneuvers and will also be used to view the surrounding lunar surface features.

Both TV displays will be of commercial quality color and frame rate with 24-inch screens. The near field resolution will provide much finer detail than the photographic map and will be used for verifying smaller than 1 meter obstacles and interesting surface features.

Because of drift in the on-board sensors and errors in following the programmed path, the predictor display can become significantly inaccurate after 15 minutes, or 1 km of travel. To correct navigation errors two techniques are available. First, the TV sensors will be used to scan for significant landmarks from which the vehicle location can be determined quite accurately. Alternately, the obstacle sensor data will be projected on the map display, corrections made to line up with known obstacles, and updated navigation data transmitted to the vehicle. Using these techniques, the system will be recalibrated periodically and the desired path maintained with significant accuracy.

Direct operator control of the vehicle will be through a joystick input to the processor. The operator will be able to control forward, rear and left-right motion with a simple four degree of freedom stick. Velocity control will be possible with the following six discrete speed positions; 8 km/hr, 4 km/hr, 2 km/hr, 1 km/hr, ½ km/hr and 0 km/hr to prevent operator uncertainty possible with a continuous velocity spectrum. These velocities will bracket the majority of allowable speeds on a variety of lunarscapes. There will be at least one full time operator and preferably an additional part time operator. One operator is needed to monitor vehicle performance and initiate navigation corrections. This same operator could handle the panning camera operation on a part time basis but a second operator trained to note unusual selenographic features is preferred. For operation in the lunar night a minimum of two lamps is suggested. One should be mounted at the front of the vehicle in continuous operation with the primary sensor and the other mounted to operate in conjunction with the panning camera particularly for hasty retreats from dangerous cul-de-sacs. Vehicle traverses at low sun angles or cross-sun to eliminate washout, glare and related problems, is further suggested. An alternate solution to these visual problems would simply be to not operate during time periods of unfavorable sun angle.

In conclusion, remote control of the MULE will be possible although significant time delay is present, by using sophisticated visual sensors coupled with one meter resolution maps and predictor display techniques. Final speed capability will depend primarily on the ruggedness of the surface to be traversed.

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15.4 Hazard Detection and Avoidance

Richard R. Pikul

The MULE is capable of going over 50 cm rocks and steps, 90 cm crevices, 30° slopes. Larger obstacles must be detected and avoided. Windows at the front of MULE provide a limited view to the operator, backed up by a television display, as indicated in Chapter 13. In remote control, television provides the visual information for driving. But as noted in the previous section, lunar optics make it difficult for the operator to see all obstacles, either directly or with TV. Obstacle avoidance sensors overcome these visual problems. In this section we discuss some technical details in designing the TV and obstacle sensors. A basic consideration is speed of the vehicle; we shall consider 10 km/hour and 2 km/hour, corresponding to the remote speeds indicated in Section 15.3 for optimum smooth surface and for rough surface.

15.4.1 Television System

Two color TV cameras provide visual information for driving near hazards. One camera is slaved to point in the direction of vehicle steering. The other points where desired by the operator.

The electronics involved is standard, except the bandwidth for two channels of commerical frame-rate, color TV may be a problem if transmitted to earth.

The slaved camera is mounted at the front of the vehicle at height H_{TV}. The vertical field of view should include the horizon to chosen $H_{TV} = 2m$, $\Theta_V = 50^{\circ}$, $\Theta_H = 65^{\circ}$.

The servo slaves the TV camera to the steering commands, so that the operator sees where the vehicle will be. The response should be quick and properly damped. Turning the MULE too quickly without seeing the new path would result in a collision with an unnoticed obstacle along side the vehicle.

The second TV camera, located at the top center of the MULE, points where the operator chooses, rotating 360° and tilting 45°. This camera provides backup to the forward camera in case of failure, provides a rear view when the vehicle travels in reverse, and permits the operator to focus on interesting surface features by zooming the field of view. As shown in Figure 15.4-2, the vehicle body blocks the camera view in the nominal configuration; the mount telescopes to give higher view angle and less blocked surface area.

15.4.2 Lighting

Camera sensors which automatically accommodate illumination levels from 2.6 lumens/m^2 (earth shine) to 1.4 10^5 lumens/m^2 (sun shine) are feasible. Artificial illumination is required for night time on the far side or to illuminate shadows at night on the near side. We do not consider artifical illumination necessary for shadows in the lunar day. Both front and rear vehicle illumination is necessary. We shall provide illumination levels at 2.6 lumens/m^2.

Specific design criteria for the lamps include the illumination pattern, intensity and angle of incidence. Because of the wash-

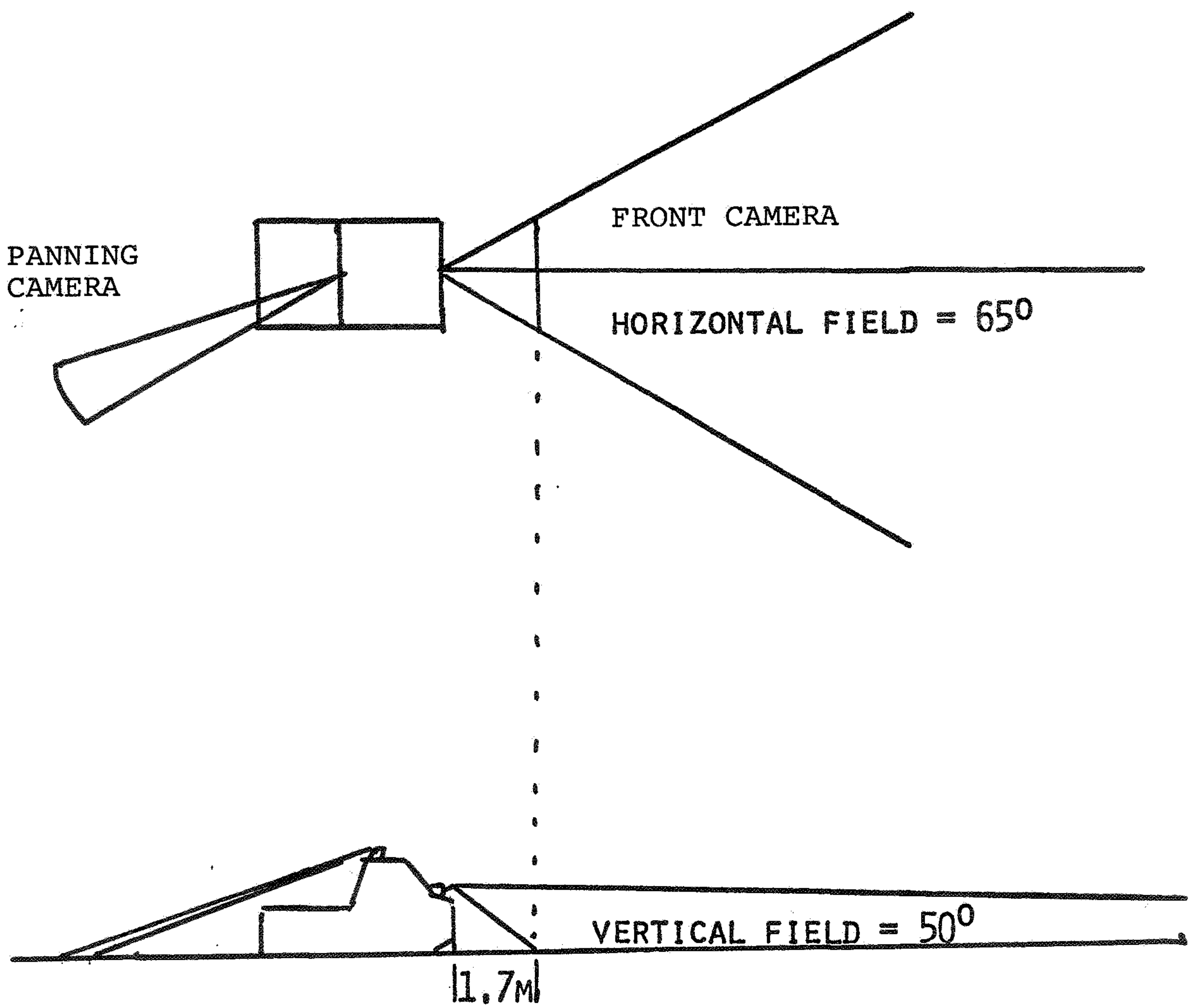

FIGURE 15.4-1 TV CAMERA FIELD OF VIEW

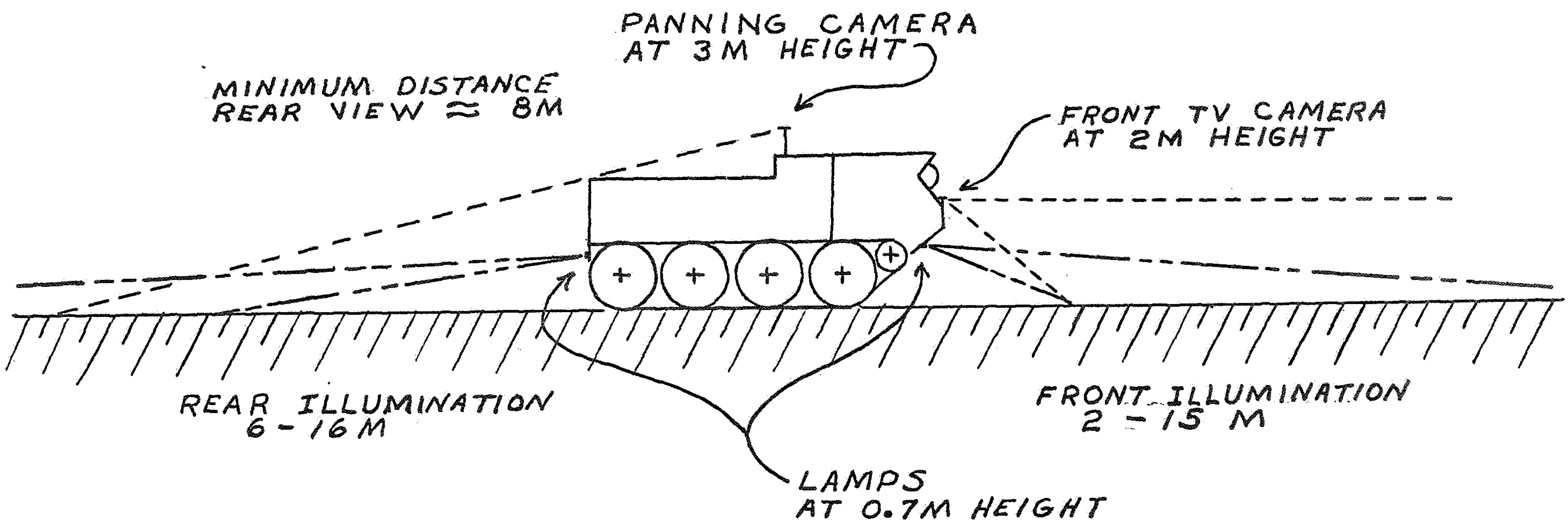

FIGURE 15.4-2 TELEVISION AND ILLUMINATION ON THE MULE

out problem the sensor view angle should differ at least 5^{o} from the illumination angle of incidence. Assuming the lamp is below the camera, the lamp should be as low as possible (sufficient to clear hazards). The maximum illumination distance, D_I, is such that

$$\cot^{-1}\left(\frac{D_I}{H_I}\right) - \cot^{-1}\left(\frac{D_I}{H_{TV}}\right) \geq 5^{o}. \tag{15.4-3}$$

With $H_{TV} = 2m$, $H_I = 0.7m$, then $D_I = 14m$. This is adequate for speeds of 5km/hour (10 second avoidance time).

The near field illumination should begin at approximately 2m, comparable to the TV near field. Likewise the horizontal pattern should be about 65^{o}.

Total luminous flux required for the pattern above (with 2.6 lumen/m^2) is about 100 lumens, requiring about 6w for each lamp.

15.4.3 Obstacle Sensors

Remote controllers depend mainly on TV pictures to control the vehicle. But lunar optics and time delays in the communications links complicate control from a remote sight, suggesting the need for additional sensor. The obstacle sensor also provides data to the driver of the manned MULE.

Large obstacles must be detected and avoided. The design of the sensor system depends strongly on the size of obstacles to be detected, plus the speed of the vehicle.

The design goal for unmanned operation is a speed of 10 km/hour;

this is based on the optimum conditions noted in Section 14.3, and contrasts with Bendix' design[2] of DLRV at 2 km/hr.

Range and heading to obstacles are required. The obvious way to obtain these parameters is by sending out a radar, infrared, or laser signal, measuring return time. The necessary size resolution and range of detection determine beamwidth: $\alpha = S/D$ in radians.

E.G., for a 0.1m resolution at 25m, the required beamwidth is 0.004 radians.

(Note: a point source signal is implied here; a radar antenna is not a point source but can be approximated as one).

Typical characteristics sensor subsystems are: (Source:DLRV)[2]

	α	Mass	Pulse width	Repetition
RADAR (0.53m dish)	0.007 rad.	3.8kg	5 nsec	780 pulse/sec
Infrared (0.05m dia.)	0.007	2.5	25 nsec	55 khz
Laser	0.003	8.2	30 nsec	55 khz

Minimum scan rate, R, is related to vehicle velocity, V, and allowable gap-lengths, L, between scans:

$$R = V/L$$

E.G., at 10 km/hr = 2.8m/sec, with 0.5m between scans, the minimum scan rate is R = 2.8/0.5 = 5.6 scans/second. The field of view (azimuth) on each scan should be sufficient to permit selection of alternate paths; e.g. three vehicle widths.

The mean time between failures (MTBF) is directly related to scan speed. The radar antenna is large and heavy; mechanical scanning failures become prohibitive at 2 scans/sec. The IR sensor is smaller, and thus more reliable. A typical laser scans electronically, and is faster. Shown below are typical figures for scanning:

	Scan rate	MTBF	
Radar	2 per sec	<1000 hr	limited by mechanical scan rate.
Infrared	2	10000	limited by mechanical scan rate.
Laser	15	4000	limited mainly by duty cycle of laser (50% assumed).

Our speed goal indicates a need for the fastest possible scan rate; as noted later, the weight penalty and MTBF penalty of laser with respect to infrared may not be significant because a single laser may do the work of two IR sensors.

Two types of warnings are suggested: a) early warning at least 10 seconds before collision to allow steering and slow-down maneuvers, and b) emergency warning about 2 seconds before collision with a command to stop. A more complete coverage of vehicle path would, of course, be desirable, but avoidance logic is much simplified by having the limited information; almost certainly, the automatic system will be backed up by the operator's visual sensing (TV if remote). At 10 km/hr, the 10 sec warning is at 28m and the 2 sec warning at 5.5m distance. Earlier warnings are possible with less reliability.

Bendix also considered "feelers" for the near field, but concluded electronic sensing was more reliable, smaller, and less weight.

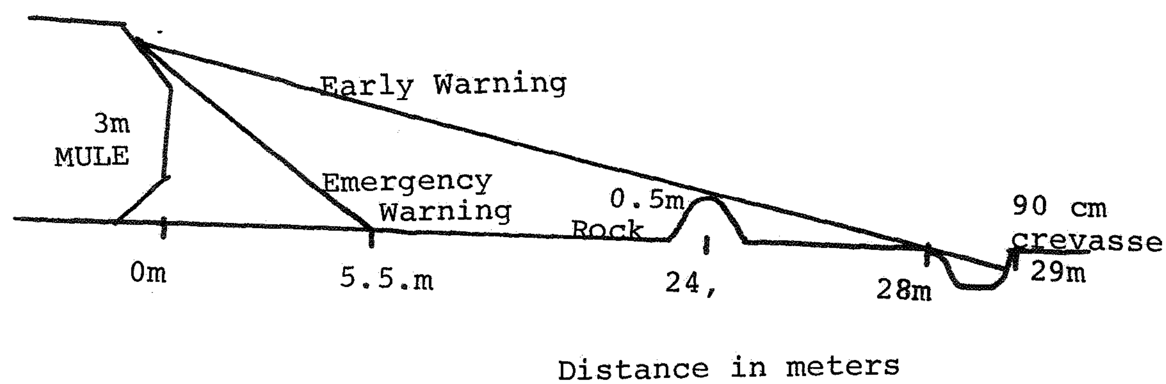

FIGURE 15.4-3 OBSTACLE SENSOR BEAMS

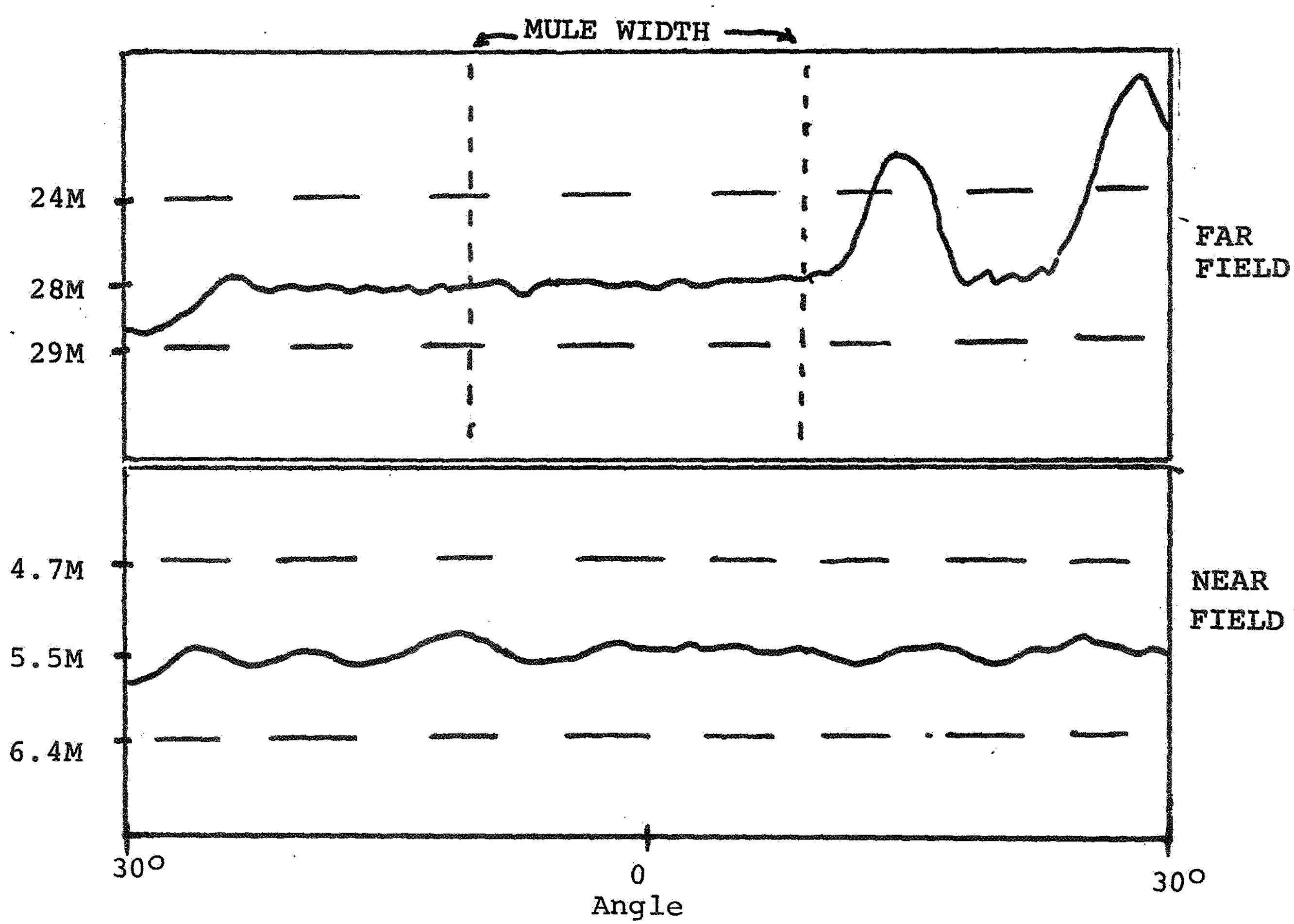

FIGURE 15.4-4 DISPLAY OF OBSTACLE SENSOR DATA

Two sensors would be required for far field and near field if radar or infrared were used. This would double the weights of the systems given above. The laser is fast enough so that it can alternate between near and far fields with the same beam; thus we recommend it.

The height of the sensor on the vehicle will influence the reliability in detecting craters, crevasses, and drops (rocks and rises are easily detected at reasonable sensor heights): in general, the higher the better. The limit on sensor height will be mainly mechanical moments, and possibly electrical links for RF data. Craters, etc. are hidden for small incidence angles.

Figure 15.4-3 shows the two beams and illustrates how distance to an obstacle (time for return of echo) corresponds to various types of obstacles. Note that crevasses, drops, etc., are the most easily missed if the scan rate is too slow or if the sensor height is too low.

The echo return can be synchronized with each pulse sent out, resulting in the display shown in Figure 15.4-4. The horizontal axis represents azimuth angle of the laser beam; the vertical represents distance (time) of the echo return. So long as the return lies inside the safe region, the vehicle need not change its path.

15.4.4 Automatic Obstacle Avoidance

On remote control missions it is desirable that the vehicle be able to make simple avoidance maneuvers when an obstacle is detected in its path. Considering the display in Figure 15.4-4, the logic to determine a hazard at a particular distance and bearing is quite simple. The reliability in avoiding false alarms is maximized by having high repetition frequency to allow several 'hits' before the obstacle is confirmed. The computer stores this hazard and keeps track of its position relative to the MULE until the avoidance maneuver is completed. The computer commands an emergency stop if the range is less then 7m (near field), or if multiple obstacles prevent simple avoidance maneuvers. Otherwise, the command is to slow down, steer left or right to avoid the obstacle, then return to the original path. In this way the preprogrammed nominal path mentioned in Section 15.3 can be accomplished without continuous attention required by the operator.

At slower speeds it may be desirable to change the ranges on the far beam and near beam. Then the display and computer would be rescaled, but the principle is the same.

Finally, the verified obstacles can be sent to the remote control station. The distance and angle to obstacles as measured by the on-board system can be compared with that shown on the photographic display. This provides a quick procedure to update the navigation computer.

15.5 Computer System

Richard Yuster

The computing facility on the MULE must be able to handle navigation and guidance requirements. In addition, it should possess the capability of on-board processing, be able to perform calculations during a traverse, monitor any experiments which may be located on board the vehicle and verify proper operations of all vehicle subsystems. There is no difficulty in selecting a computer to fullfill the above requirements. However weight, power consumption and size impose additional constraints which must be met in the design.

The navigation and guidance computer on board the Apollo spacecraft could provide the necessary computing capability for this lunar mobility system. Listed below are the important characteristics of the navigation and guidance computer aboard the Apollo.

Volume:	1.0 ft^3
Weight:	58 lb.
Power Consumption:	100 watts
Word Length:	16 Bits
Add Time:	23.4 M Sec.

A microprocessor is presently being developed for the 1980-1982 space shuttle. This computer is composed of 3 major components, a processor, a fixed and variable memory and input/output units. The microprocessor utilizes state of the art technology featuring integrated circuits and flexible internal command structure.

Listed below are the important characteristics of the microprocessor.

Volume:	1 cubic inch
Weight:	1 ounce
Power Consumption:	2.5 Watts
Word Length:	16 Bits
Add Time:	20 M Sec

It is apparent from the above data that small, light weight computer systems with low power requirements will be developed and available for the MULE.

REFERENCES

Hopkins, A.L., "Electronic Navigation Charts Man's Path to the Moon", Space Electronics, January 9, 1967

15.6 Summary

G. Lawrence

Figure 15.6-1 illustrates the astrionics subsystems on-board the MULE, with major interface connections. Signals communicated outside the vehicle are omitted.

Figure 15.6-2 illustrates appropriate subsystems at the remote stations.

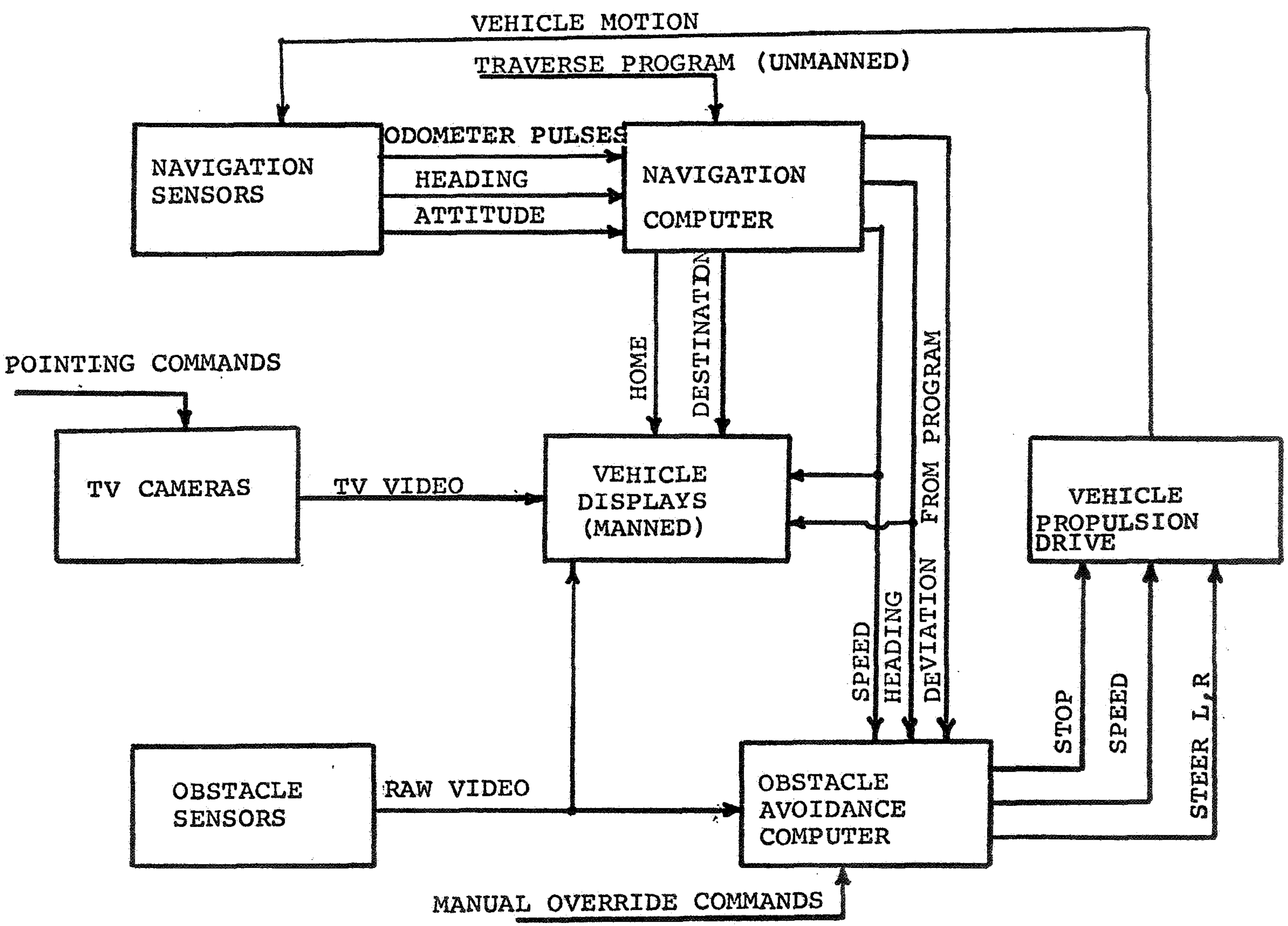

FIGURE 15.6-1 ASTRIONICS SUBSYSTEMS

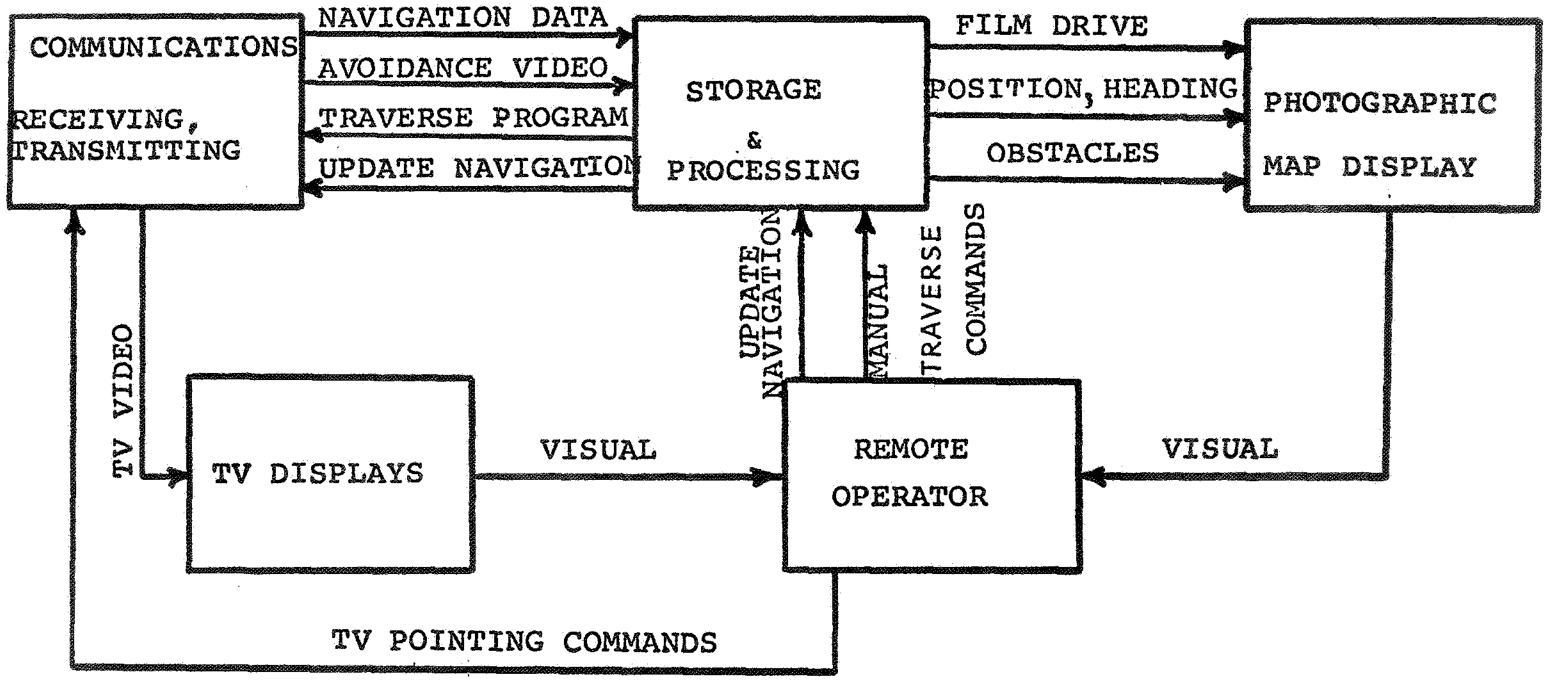

FIGURE 15.6-2 REMOTE CONTROL STATION

PART IV

SUPPORTING REQUIREMENTS

This part of the report discusses the more obvious supporting requirements for the MULE. It also includes chapters on thermal analysis of the system and a satellite communication system.

CHAPTER 16

FAILURE EFFECTS ANALYSIS

F. J. Kay
R. Ray Nachlinger

As was specified in the statement of work, the vehicle was designed so that it could still operate if it had a failure in a single subsystem. This necessitated considerations such as drive wheels that could be used if we had a track failure. In addition, since each wheel is driven by a separate motor, the vehicle will still operate with the failure of one motor.

This philosophy was continued throughout the design. We thus have both a cabin and suit for life support, two communication systems, and three sources of power. The preliminary design of the vehicle appears to be immune to single point failures.

The biggest danger to the success of a mission appears to be from factors over which we have no control, such as operator errors and natural lunar hazards. These factors were minimized by generous clearances and stability characterics and by an obstacle avoidance warning system. It must be emphasized that this danger was only minimized, since nothing can absolutely eliminate the chance of an operator error. The backup for this type of failure comes from a reserve space tug which can be dispatched to any spot on the lunar surface to rescue stranded astronauts.

In conclusion, the greatest danger to the crew appears to be from impact during a collision or roll of the vehicle.

CHAPTER 17

RESEARCH AND DEVELOPMENT REQUIREMENTS

F. J. Kay
R. Ray Nachlinger

Since the vehicle we designed was to be operational in the 1980's, we assumed that there would be advances in several areas before that time. There are five areas of research and development that should receive considerable attention before an efficient vehicle can be designed to operate in the hostile lunar environment. These are tracks, friction reduction, energy dissipation, batteries, and control.

As with any track system, the major problem is finding a material that is strong enough, flexible enough, and resistant to fatigue failure. The lunar environment complicates the problem by the large range of temperatures at which the tracks must operate. ($-250^{\circ}F$ to $250^{\circ}F$). We know of no non-metalic material that has "nice" properties throughout this range of temperatures.

The second area that needs to be looked at is friction reduction. This includes such problems as cold welding, bearing design, lubrication, and dust control. The problems here are not only the extreme temperatures, but also the low pressures and the lack of convection for cooling. These problems are the same that are involved with any energy dissipation device.

Problems affiliated with dust control need to be studied closely. Since dust affects the visibility and remote control capabilities it becomes quite important. Bearings and other critical areas that would be affected by dust particles must be isolated and protected from them.

There are also some problems which do not depend on the hostile environment. These include continuous communication, high resolution mapping, and perhaps most interesting, time delayed remote control. This problem stems from the communication delay from the earth to the moon. With this three second delay, it appears that continuous remote control would be impossible.

In the 1980 periods we believe that materials will be available that will allow us to decrease the weight of the structure and, thus, lower the weight of the vehicle. Materials should be developed that are light in weight and high in strength. Possible materials are beryllium alloys, sandwich materials, perhaps even high strength irradiated polymers will be available. This area is one that should be researched thoroughly by the aerospace industry. Temperature fluctuation causes thermal cycling and thermal shock which place severe limitations on the material.

Other suggested areas of research have been incorporated in the chapters on astrionics and human factors.

CHAPTER 18

CREW TRAINING

George V. Treischmann

Crew training for manned operation of the proposed tracked MULE is not a critical consideration.

Experience with crew training for manned operation of the Apollo Lunar Rover vehicle[1] indicates that a ten hour training syllabus is adequate to produce a competent vehicle operator from a qualified astronaut. This ten hour syllabus includes introduction to the mechanical systems of the Lunar Rover and operational training in simulated 1/6 G Lunar environment.

The MULE operator will have more complicated electronic and mechanical systems to deal with, and familiarization in these systems will, therefore, increase training time. This increase has been estimated by the Human Factors group of the NASA-ASEE System Design Team to be three-fold, thus requiring a thirty hour syllabus. A general summary[2] of training procedures for manned operation of the MULE are given in Table 18-1.

Training for remote operation is perhaps the most significant training problem raised by the MULE concept. Assuming an earth stationed operator of an unmanned Lunar vehicle presents a need for remote operation with a time delay. This problem is covered in the Remote Control section of this report.

TABLE - 18-1

PROCEDURES TRAINING

- PREPARE STUDY MATERIAL & FAMILIARIZATION DATA FOR CREW
- ARRANGE FOR CREW SCHEDULING THROUGH TRAINING COORDINATOR
- PREPARE SIMULATION/MOCKUP AREA & SETUP FOR EXERCISE
- ARRANGE FOR SUIT, LIFE SUPPORT, COMMUNICATION, PHOTOGRAPHIC AND MOCKUP/TRAINER SUPPORT
- COORDINATE SUPPORT BY PROJECT ENGINEERS, PRINCIPAL INVESTIGATORS, CONTRACTORS FOR LUNAR SURFACE EQUIPMENT AS REQUIRED
- PREBRIEF CREW
- CONDUCT OR ASSIST IN CONDUCT OF EXERCISE
    - TIME PROCEDURES, INSTRUCT CREW, ACT AS COMMUNICATOR, DIRECT TECHNICIANS, DIRECT SUPPORT
- DEBRIEF CREW & ASSIGN ACTION ITEMS
- DOCUMENT TRAINING EXERCISE

---

[1] Koppa, R. J., General Electric Co., presentation to Human Factors Section of NASA and ASEE Summer Systems Engineering Design, Faculty Fellowship Program. July, 1970. Ellington AFB, Houston, Texas.

[2] Koppa, R. J., "Flight Crew Procedures Development and Training for Lunar Surface Operations presented at the Southwestern Psychological Association at St. Louis, Mo. April 23, 1970.

CHAPTER 19

SATELLITE SYSTEM

Richard Yuster

Part of the lunar program portion of the manned space flight integrated plan is to systematically explore both sides of the moon with the aid of both manned and automated mobility vehicles. These lunar roving vehicles will be monitored and/or controlled primarily from the surface lander or surface bases established on the moon. Personnel located on the lunar orbiting space station may also assist when the mission requires an unmanned traverse of long duration. Since unmanned operation of a mobility aid requires continuous communications, a major problem arises when the vehicle approaches the far side of the moon. Effectively, the communications link breaks down when operations on the far side of the moon are considered. Thus communications with a far side terminal, or an orbiting spacecraft hidden by the moon will involve some form of intermediate relay. The existing flights of the Apollo program severely suffer due to the loss in communication, when the moon moves between the spacecraft and the earth. Critical maneuvers occurring behind the moon (such as SOS ignition, emergency landing of the LEM on the far side, a possible rendevous between vehicles) already exemplify the apparent need for communication links to be established. This chapter will explore possible alternatives to the problem of providing continuous communication coverage on the moon.

19.1 Alternative Communication Configurations

It is possible to achieve adequate communication coverage by employing several different techniques. However, some configurations will prove to be too costly and will be rejected without further mention.

19.1.1 Lunar Surface Links.

One approach is to erect relay stations along the surface of the moon. The height and distance between towers can easily be determined by a simple analysis which follows. Figure 19.1-1 illustrates the geometry involved in the computation.

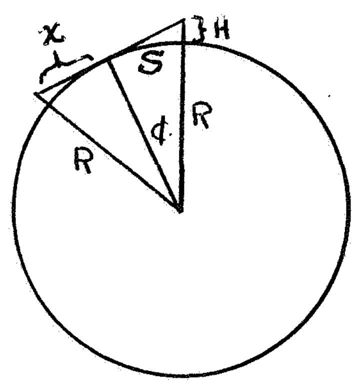

FIGURE 19.1-1 GEOMETRY FOR LINE OF SIGHT COMPUTATIONS

Referring to the diagram, some simple relations will be listed without proof:

$$\phi = \frac{S}{R} \tag{19.1-1}$$

$$x = (H+R)\ \sin\ \phi \tag{19.1-2}$$

Applying Pythagorean Theorem

$$x^2+R^2=(H+R)^2 \quad (19.1\text{-}3)$$

Simplifying and substituting Equation 19.1-3 becomes

$$H^2+2RH-R^2 \operatorname{Tan}^2 \phi = 0 \quad (19.1\text{-}4)$$

Solving for H and simplifying gives the desired relation between the height and distance between antennas.

$$H=R\left(\frac{1}{\cos \phi} -1\right) \quad (19.1\text{-}5)$$

Thus, since ϕ is extremely small, $\cos \phi$ may be expanded in a Taylor series

then $\cos \phi = 1- \frac{\phi^2}{2!} + \frac{\phi^2}{4!}\ldots$ (19.1-6)

$$\cos (S/R) = 1-\frac{(S/R)^2}{2!} + \frac{(S/R)^4}{4!} \ldots \quad (19.1\text{-}7)$$

As a sample calculation, suppose the ration of $S/R = 0.01$ substituting into Equation (19.1-5) results in an antenna height of 290 ft. for a maximum distance between relay stations of 21.6 miles. Thus, this is a highly impractical mode of transmission.

Another scheme suggests the laying of hard wire between surface terminals. Figure 19.1-2 is a plot of weight versus distance to deploy copper wire 0.01 inches in diameter. A typical mission of 1500 km. requires 5,280 lb. of copper which weighs about half as much as the mobility aid. As a final alternative, a surface wave transmission link was suggested, but the band width requirements limit operations to the low frequency region of the spectrum.

19.1-2 Libration Point Satellites

There are points within our planetary system where the forces are exactly balanced, i.e. a body located at any of these points would experience no resultant force. For example, an isolated two body system (earth-moon) has within its gravitational field five points where a satellite placed with proper velocity and direction will fly in formation with it. At these "libration points", as they are frequently called, the centrifugal force will exactly balance the gravitational ones. Figure 19.1-3 illustrates the five libration points associated with the earth-moon system. Deploying a satellite in the vicinity of a libration point requires little thrust and therefore, fuel to maintain its position. In addition, this may be facilitated by present satellite translational control systems. Since the L_2 stagnation point lies behind the moon, a satellite placed properly near here could provide the needed relay link between the backside of the moon and the earth. In addition, as part of a lunar relay network, it would complete a surface link, i.e. communications between any two points on the lunar surface. The possibilities, although not exhaustive, exhibit the importance and consideration which should be given to this method. There are several possible schemes that have been proposed which utilize the L_2 libration point. Among the most favorable and practical are the Hummingbird and Halo.

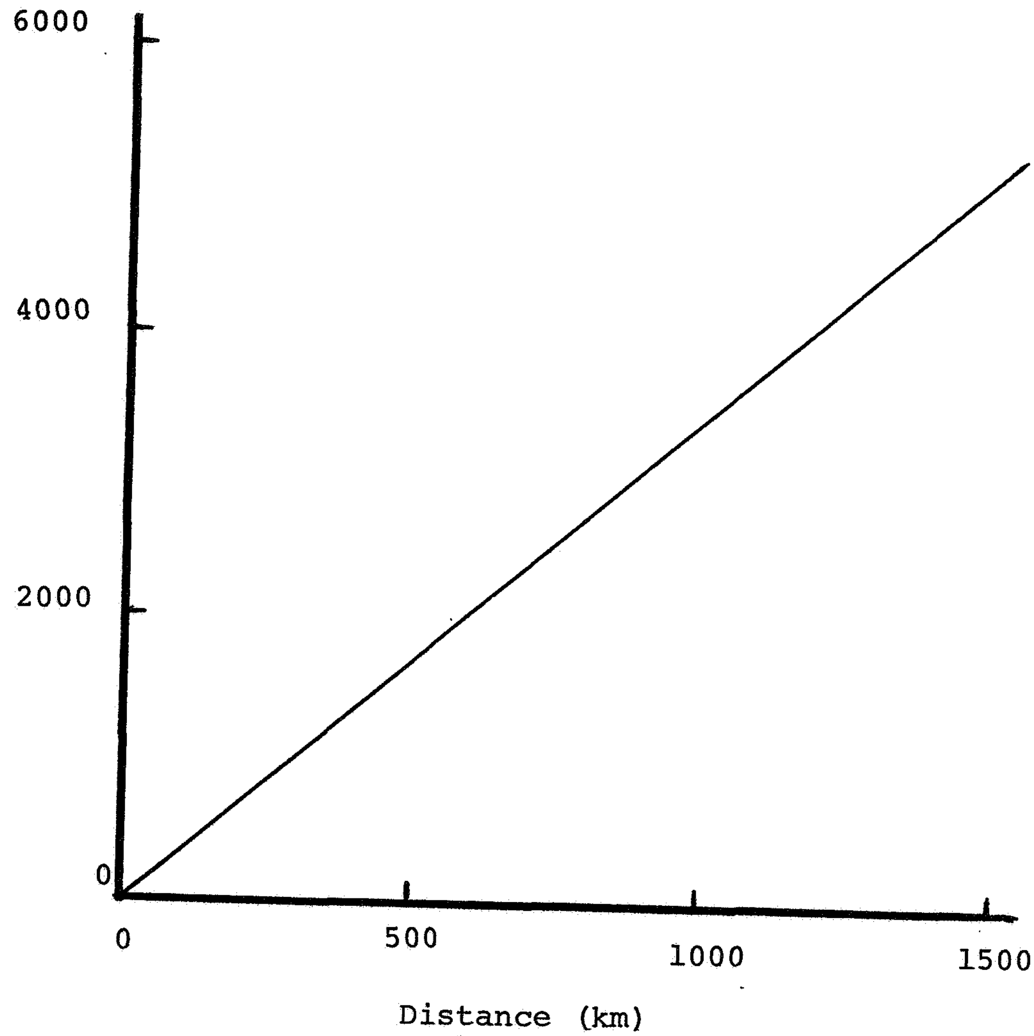

FIGURE 19.1-2 WEIGHT vs DISTANCE TO DEPLOY 0.10 INCH DIAMETER COPPER WIRE

19.1.2.1 Hummingbird Libration Relay Satellite

The Hummingbird libration satellite or "Hovercraft", as it is sometimes referred to, maintains a fixed position with respect to the earth-moon system as illustrated in Figure 19.1-4. This is accomplished through continuous low level thrusting. The minimum height which must be maintained by the satellite in order to communicate past the edge of the moon can easily be derived through the use of geometric techniques. Such calculations show a libration satellite must maintain at least an altitude h of 3100 km for an unobstructed view of the earth. To guard against possible occulations by the moon an additional 200 km should be added to the minimum altitude and this will be reflected in all further calculations. The Hummingbird can be placed optimumly along either the vertical or orbital axis to minimize the amount of acceleration on the satellite. Figure 19.1-5 which was derived by utilizing the equation of motion for a three body system, gives a graphical account of the acceleration encountered as a function of distance from the libration along either the vertical or orbital axes. Figure 19.1-6 illustrates the possible locations for placing a Hummingbird satellite with an altitude of 3300 km and its associated acceleration. Obviously, it would cost less in fuel consumption to deploy the satellite along the orbital axis. To maintain proper position, a low thrust high impulse engine should be employed for the Humminbgird configuration. Although continuous thrusting is required, the daily fuel consumed will be small.

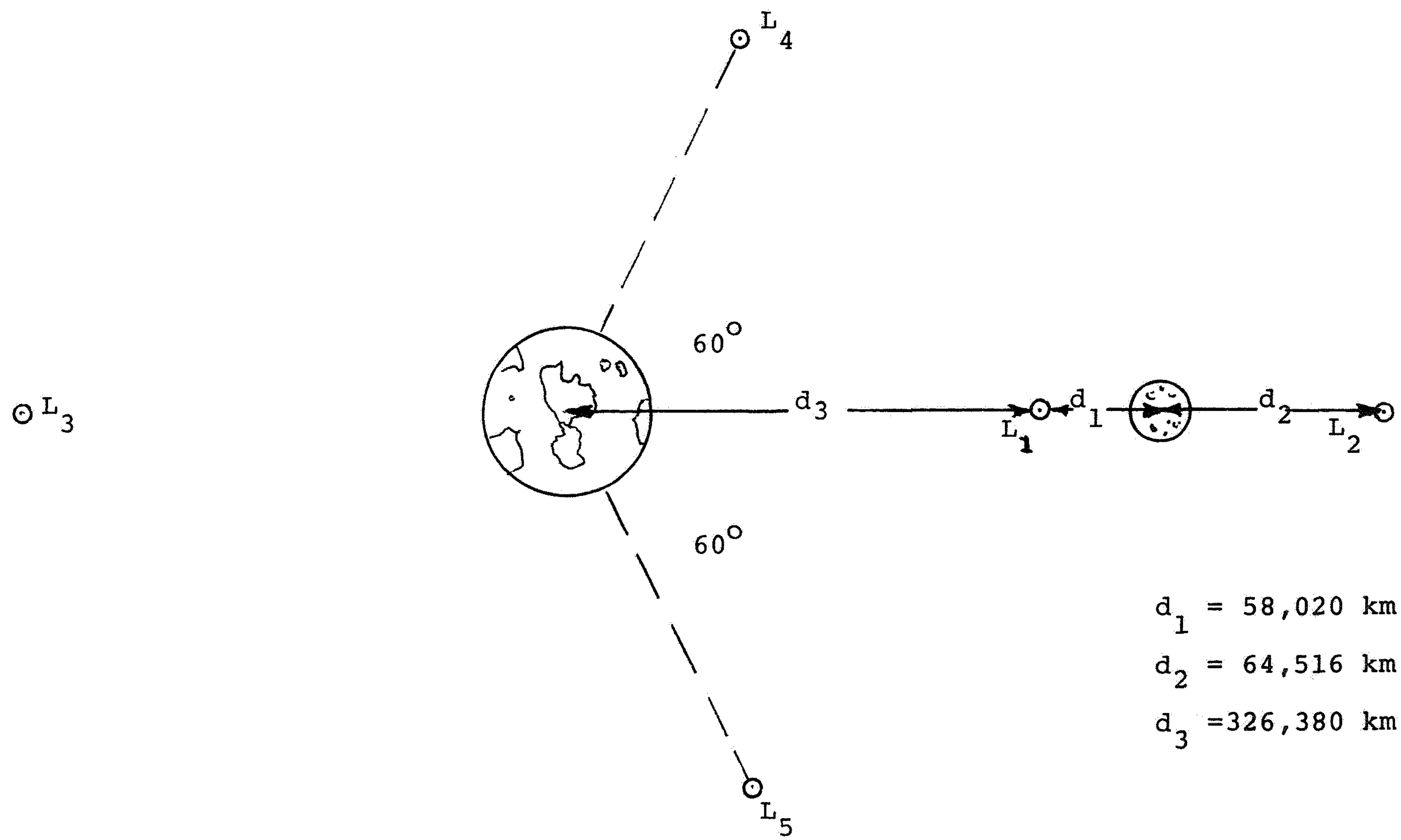

FIGURE 19.1-3 LIBRATION POINTS ASSOCIATED WITH A TWO BODY SYSTEM

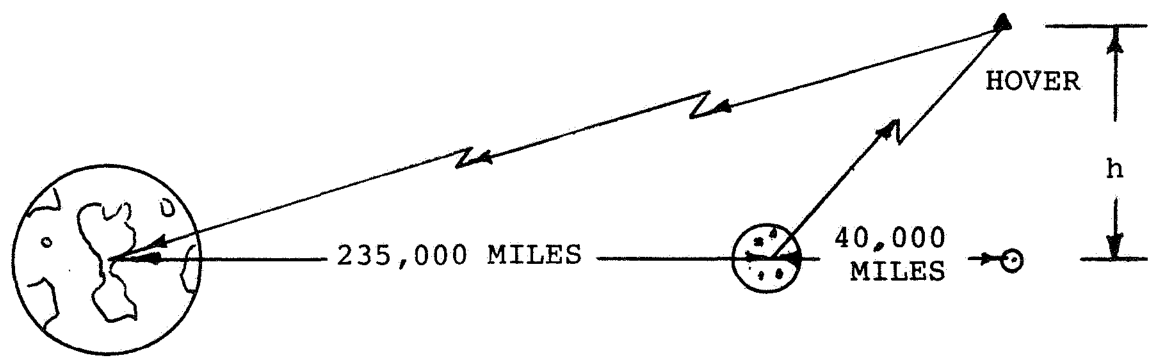

FIGURE 19.1-4 HUMMINGBIRD LIBRATION RELAY SATELLITE

19.1.2-2 Halo Libration Relay Satellite

Another concept which takes advantage of the libration point is the "halo" satellite. In this configuration orbit about the L_2 libration point. This is in a plane perpendicular to the moon's orbital plane. Selection of the orbit is made so that the satellite is in constant view of the earth while it rotates like a halo about the moon. Figure 19.1-7 illustrates the geometry involved in backside communications only. Since this type of relay is revolving in a near circular orbit the average acceleration for a 3300 km orbit is approximately 4.94×10^{-6} m/sec^2. This assumes that the method for controlling the oscillations is single axis Z control. Thus it can be observed that the cost in deploying a halo is smaller than for the Hummingbird by at least one order of magnitude. A more detailed analysis, carried out in the flight dynamics section of the report will verify this. Unlike the Hummingbird, a high thrust low impulse engine would be more appropriate for the trim maneuvers involved.

19.1.3 Lunar Orbiting Satellites

Unlike the earth, there is no stable synchronous orbit about the moon. Any satellite placed in orbit will revolve with respect to the moon. Therefore, a relay placed in lunar orbit will suffer communication blackouts, due to occasional occultation of the satellite from the earth by the moon. The frequency differs depending on the orbit chosen. Figure 19.1-8 illustrates an equatorial system of satellites, i.e. satellites deployed

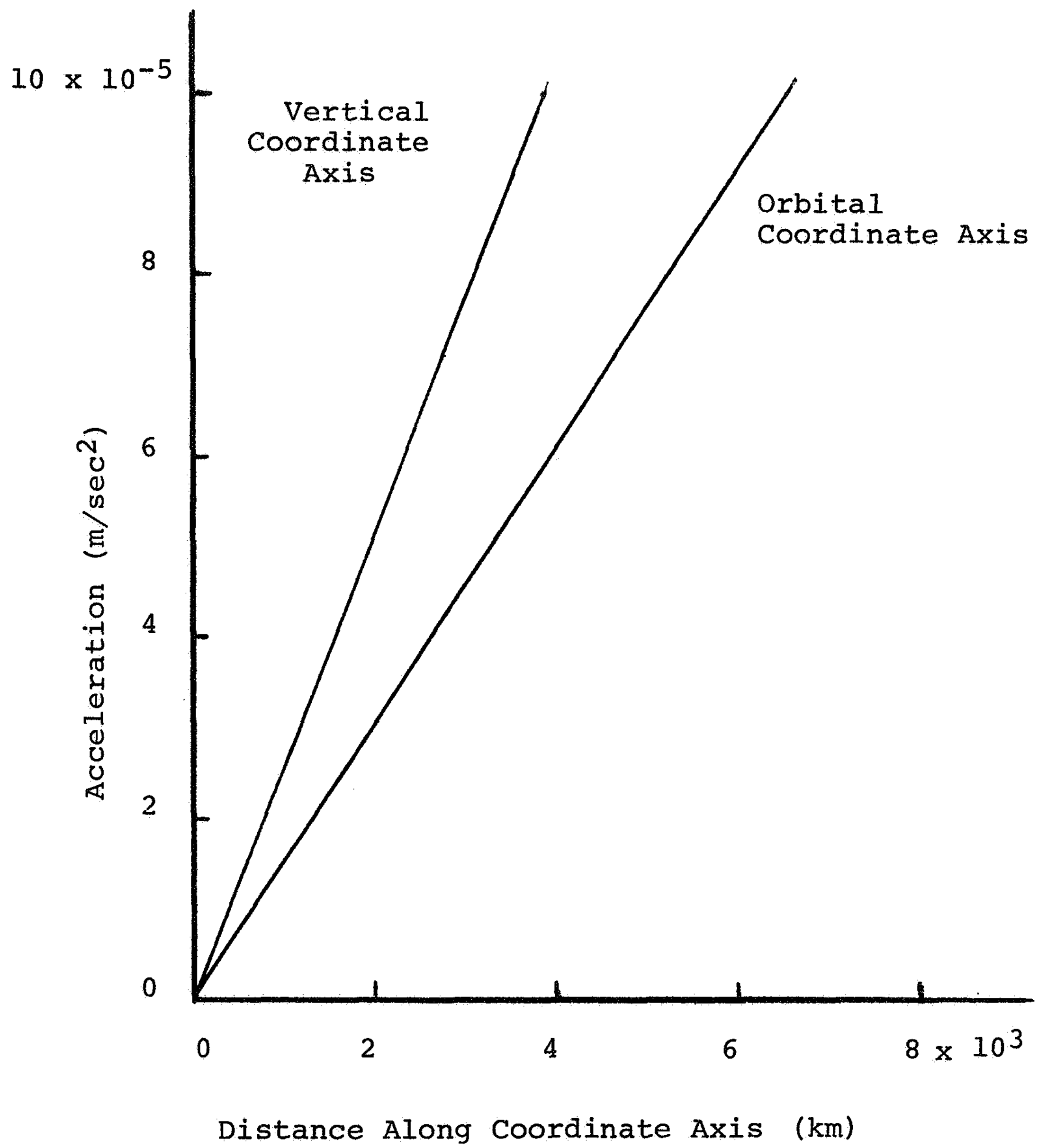

FIGURE 19.1-5 ACCELERATION vs DISTANCE ALONG COORDINATE AXIS

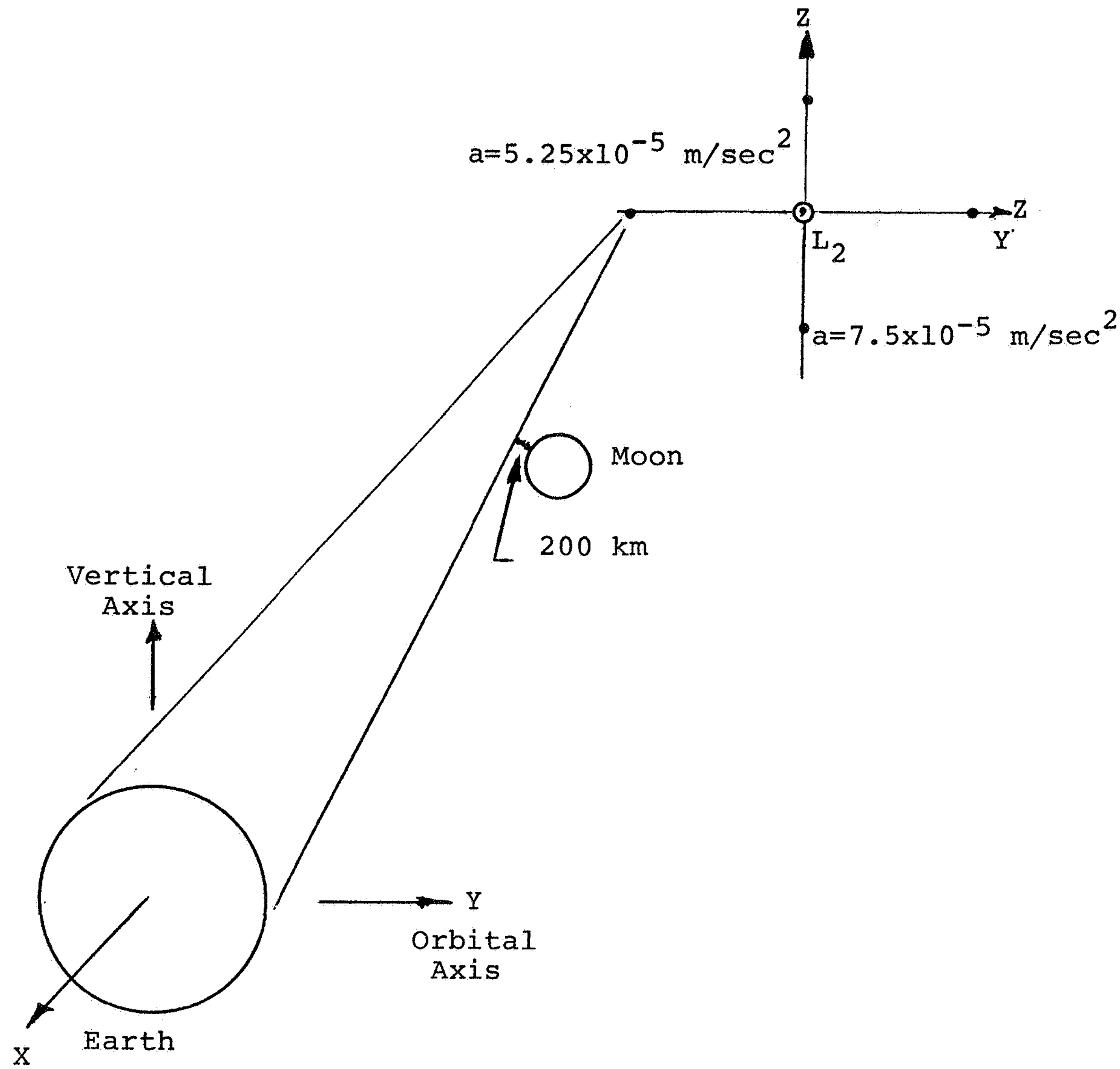

FIGURE 19.1-6 POSSIBLE LOCATIONS FOR PLACING THE HUMMINGBIRD LIBRATION RELAY SATELLITE

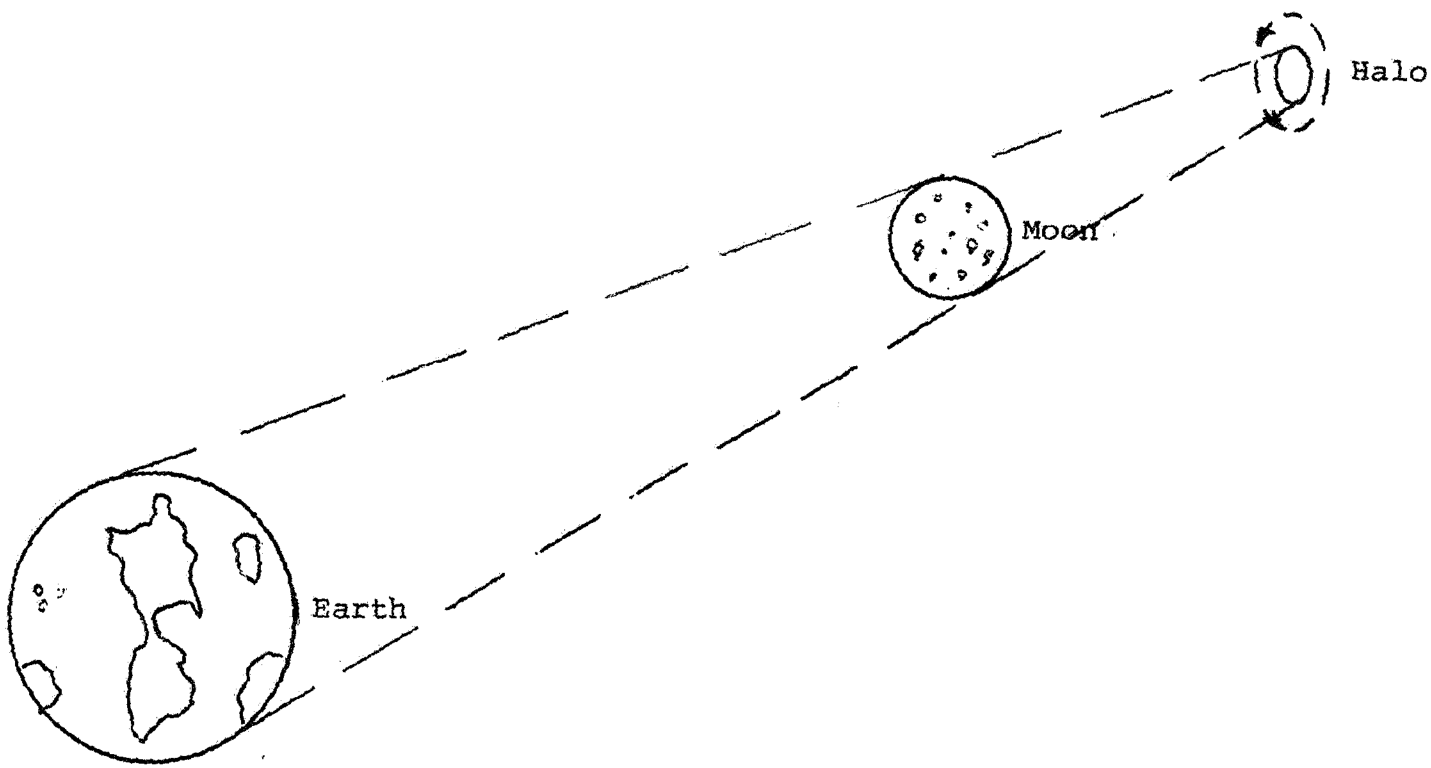

FIGURE 19.1-7 HALO LIBRATION RELAY SATELLITE

about the equator of the moon. The equatorial system of satellites requires a minimum of five satellites equally phased for adequate coverage in the orbital plane. This places the constraint that all missions must be restricted to within the moons orbital plane. A direct violation of the integrated plan, which calls for exploration of the polar caps. In addition, a basic limitation to this technique is that each satellite is occulted by the moon once every revolution. Due to the low frequency of occultation each month and the capability of accessing almost any point on the surface of the moon, the polar orbit is superior to the surface of the moon, the polar orbit is superior to the equatorial orbit. Figure 19.1-9 illustrates a system of polar satellites about the moon. If a continuous communications link is desired between earth and a lunar base located at any point on the surface of the moon, a total of 9 satellites should be deployed in polar orbits. This would require three relay satellites per polar orbit and each orbit 60^{o} out of phase with the other.

19.2 Flight Dynamics

Flight dynamics is primarily concerned with two categories; A) the establishment of satellites in proper orbit, B) housekeeping chores of keeping them there. A comparison is performed to determine the cost involved in deploying and maintaining a communications relay satellite in lunar orbit versus one about the earth-moon libration point - L_2. In all cases it is assumed that the transfer will occur from an earth parking orbit

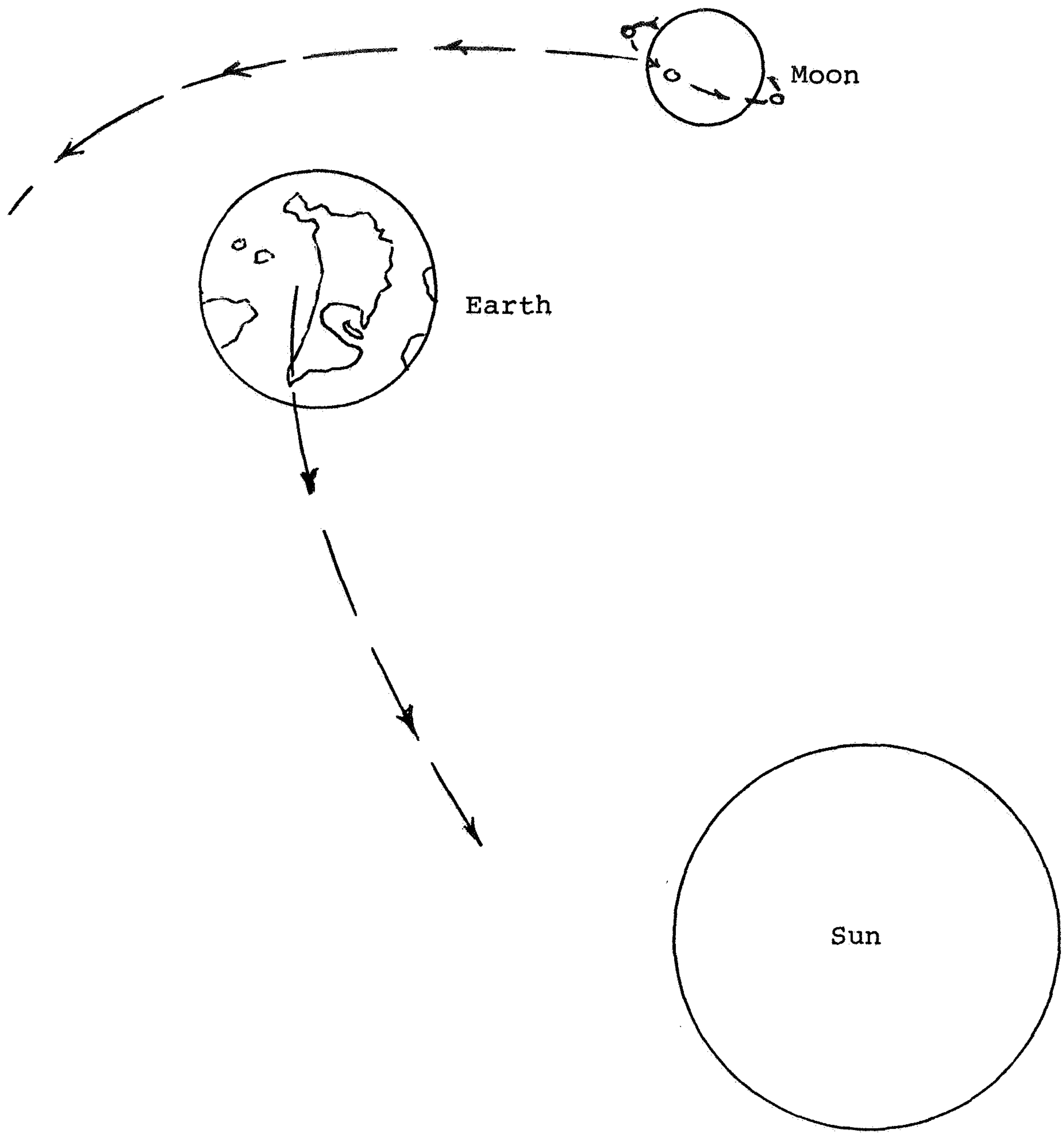

FIGURE 19.1-8 EQUATORIAL RELAY SATELLITE SYSTEM

whose altitude is approximately 100 nautical miles. A launch vehicle fired at Cape Kennedy will initially place the satellite system in this orbit. Three relay satellite systems were compared:

A. Halo
B. Hummingbird
C. Polar

19.2.1. Deployment

19.2.1.1. Halo and Hummingbird

Although the Halo and Hummingbird utilize different schemes about the libration point, both employ the same transfer trajectories. Therefore, the discussion which follows will pertain to both concepts.

An infinite number of transfer trajectories between an earth parking orbit and L_2 orbit are possible. However, various constraints such as flight times, Δv requirements, and the number of possible launch windows per month reduce this quantity if only practical missions are to be considered. In particular Figure 19.2-1 illustrates the types of transfers of any kind are deemed too costly since their Δv requirement exceeds the fly-by mode by a factor of four. The indirect mode of transfer which utilizes the moon to lower the velocity requirements falls into two categories - fast and slow lunar fly-by transfers. The slow transfer has the advantage of never having its vehicle being occulted by the moon, i.e. all operations are performed in full view of the earth. The fast mode of transfer although undergoing a period of occultation by the moon requires slightly less Δv.

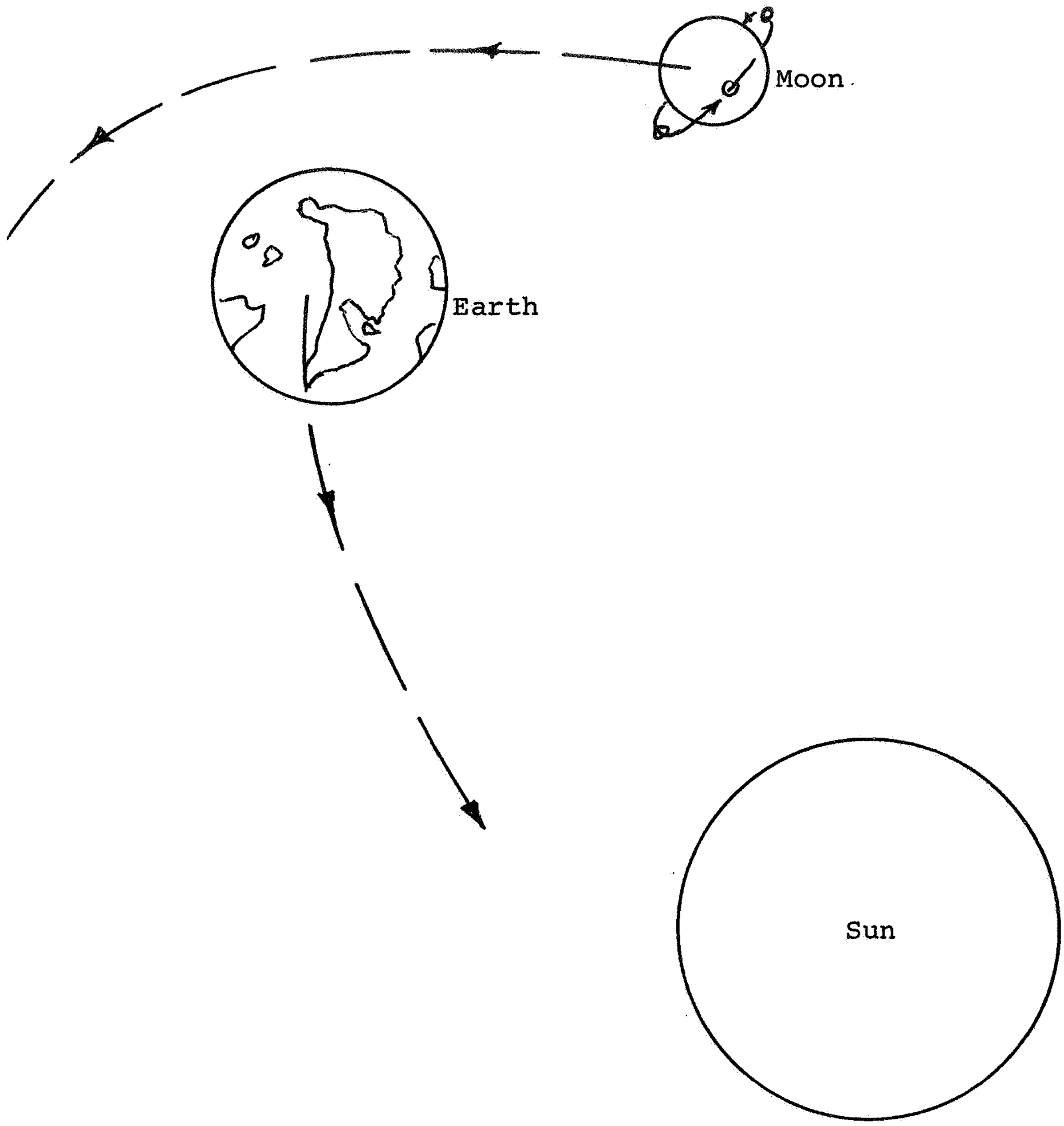

FIGURE 19.1-9 POLAR RELAY SATELLITE SYSTEM

Representative values are enumerated in Table 19.2-1 below. The various transfers considered without midcourse corrections are included.

TABLE 19.2-1 HALO SATELLITE ENERGY REQUIREMENT

Trajectory	Flight Time	Δv (M/sec)
Direct Transfer	4 days	1230
Indirect Slow Fly-By	17.86 days	353
Indirect Fast Fly-By	8.59 days	333

Obviously, an indirect-fast lunar fly-by trajectory mode is preferred. Table 19.2-2 gives, in detail, the propulsion requirements for the fast fly-by trajectory used to deploy a satellite near the L_2 libration point.

TABLE 19.2-2 VELOCITY IMPULSE REQUIREMENTS FOR DEPLOYMENT OF LIBRATION POINT (L_2) SATELLITE

Maneuver	Δv (M/sec)
Earth-Moon Midcourse Correction	37
Velocity Impulse Near Moon	191
Moon-L_2 Midcourse Correction	6*
Orbit Establishment About L_2	142**
Total	376

*Fast Lunar Fly-By Trajectory.

**Halo and Hummingbird are essentially the same.

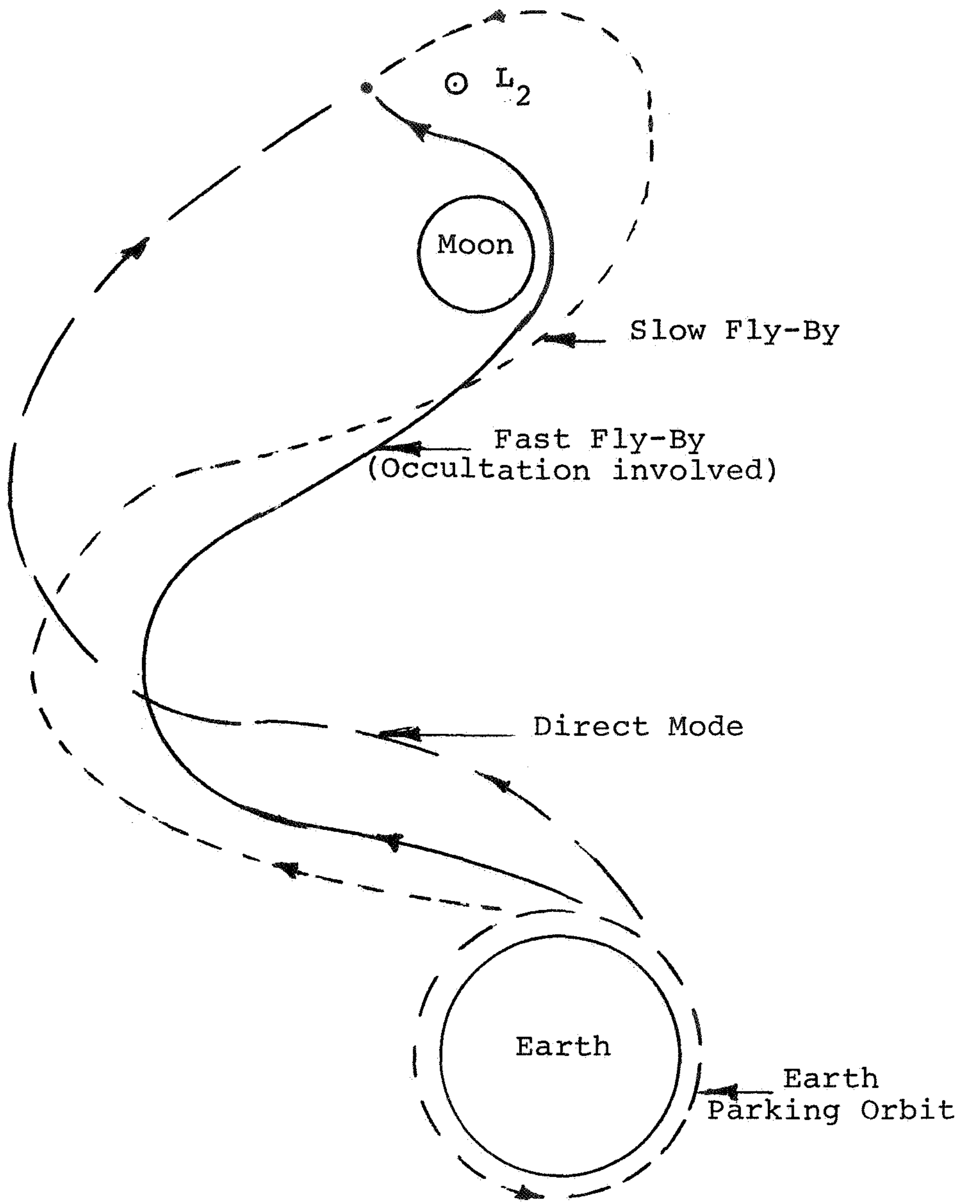

FIGURE 19.2-1 TRANSFER TRAJECTORIES

Sun

19.2.1.2 Polar Satellite

The impulse velocity required to place a satellite in lunar orbit is a function of three parameters: 1. Flight time between trans-lunar injection from an earth parking orbit and lunar orbit insertion. 2. The moon's orbital plane.

Table 19.2-3 illustrates the relation between the parameters described above for three selected altitudes; 2000, 6000 and 10,000 nautical miles. Interpolation techniques can be performed on the values enumerated in the table to obtain data for other altitudes. It should be noted that the velocity requirements are essentially independent of orbit altitude. As the flight times increase, a smaller Δv will occur at maximum distance at zero degrees inclination. Thus it may be concluded that the velocity impulse required at lunar orbit insertion will decrease as: 1) The flight time is increased. 2) The distance between the earth and moon approaches the maximum value 432,000 km. 3) The outbound inclination angle approaches 0°. Finally in all cases, it should be noted that the cost in deploying a satellite into lunar orbit (polar or equatorial) results in a larger velocity impulse than that required by a libration satellite. Thus the average cost in Δv to deploy a satellite into polar orbit (6000 nautical miles) about the moon would be approximately 780 m/sec.

19.2.2 Station Keeping

Once the satellites are placed in their proper orbit, there remains the task of periodically performing trim maneuvers to keep them on station. For example, a halo satellite if permitted to go without any orbit correction would remain in an orbit about the L_2 point for some months, without being occulted by the moon. The Hummingbird satellite, however, would be in a noncommunicable position after only 10 days had elapsed. The polar satellite depending on its altitude would probably remain useful for about four to six months after thrust has been lost.

19.2.1 Hummingbird

From Newton's 2nd law, if an object is to remain in equilibrium the vector sum of the forces exerted on that object must be equal to zero. Therefore, the Hummingbird relay satellite, in order to remain stationary with respect to the earth-moon system, must produce a force equal in magnitude and opposite in direction to the acceleration exhibited at the point in space where the craft is located. Thus, the thrust required by the on-board stabilizer engine is given by Equation (19.2-1).

$$\frac{d(M\bar{v})}{dt} = \bar{f}_{Eng} = m\bar{a}$$

Where $\bar{f}_{Eng}$ - force produced by the on-board stabilizer engine

m - mass of the satellite which is a fraction of time

v - exhaust velocity

a - the acceleration which the Hummingbird must overcome.

Expanding Equation (19.2-1)

TABLE 19.2-3 IMPULSE VELOCITY REQUIRED FOR LUNAR ORBIT

Where g is earth gravity

Finally $$m(t)\ m_i e^{-\left(\frac{at}{gI_{sp}}\right)} \qquad (19.2\text{-}6)$$

Orbit - 2000 Nautical miles

Time (hr)	Inclination angle (0)	Distance To Moon	Δv (M/Sec)
60	60	max	1,020
60	0	max	975
60	60	min	905
60	0	min	837
80	60	max	760
80	0	max	700
80	60	min	740
80	0	min	677
110	60	max	670
110	0	max	610
110	60	min	716
110	0	min	640
	Orbit - 6000 Nautical miles		
60	60	max	1,070
60	0	max	1,005
60	60	min	936
60	0	min	870
80	60	max	778
80	0	max	703
80	60	min	740
80	0	min	664
110	60	max	657
110	0	max	555
110	60	min	717
110	0	min	618

TABLE 19.2-3 (cont'd)

Orbit - 10,000 Nautical miles

60	60	max	1,120
60	0	max	1,060
60	60	min	963
60	0	min	870
80	0	max	790
80	0	max	702
80	0	min	747
80	0	min	655
110	60	max	640
110	0	max	550
110	60	min	717
110	0	min	610

$$m \frac{d\bar{v}}{dt} + v\frac{dm}{dt} = -m\bar{a} \qquad (19.2\text{-}2)$$

Since the operation is continuous in nature, $\frac{d\bar{v}}{dt}$ is equal to zero.

Thus
$$v \frac{dm}{dt} = m\bar{a} \qquad (19.2\text{-}3)$$

Integrating by parts results in the following expression for the mass of the satellite as a function of time t.

$$m(t)=m_i e^{-(\frac{a}{v})t}$$

Where m_i=total of satellite mass at time zero.

The specific impulse, I_{sp} which is a measure of how effective a propulsion system uses its supply of fuel can be related to the exhaust velocity. This reaction is

$$I_{sp} = \frac{v}{g} \qquad (19.2\text{-}5)$$

Where g is earth gravity.

This expression will give us the important relation between the life time of the satellite, and the remaining mass for a particular type engine. In fact, if Equation (19.2-6) is normalized with respect to the mass at launch and subtracted from Equation (19.2-1), the result would be the percentage of fuel to the total mass consumed in time "t" for station keeping. The graphic results are indicated in Figure 19.2-6.

To illustrate the use of Figure 19.2-6, suppose the on-board stabilizer engine develops a specific impulse of 400 sec and is required to have a life time of three years, then the corresponding amount of fuel consumed in station keeping would be 12% of the total oribinal mass of the relay.

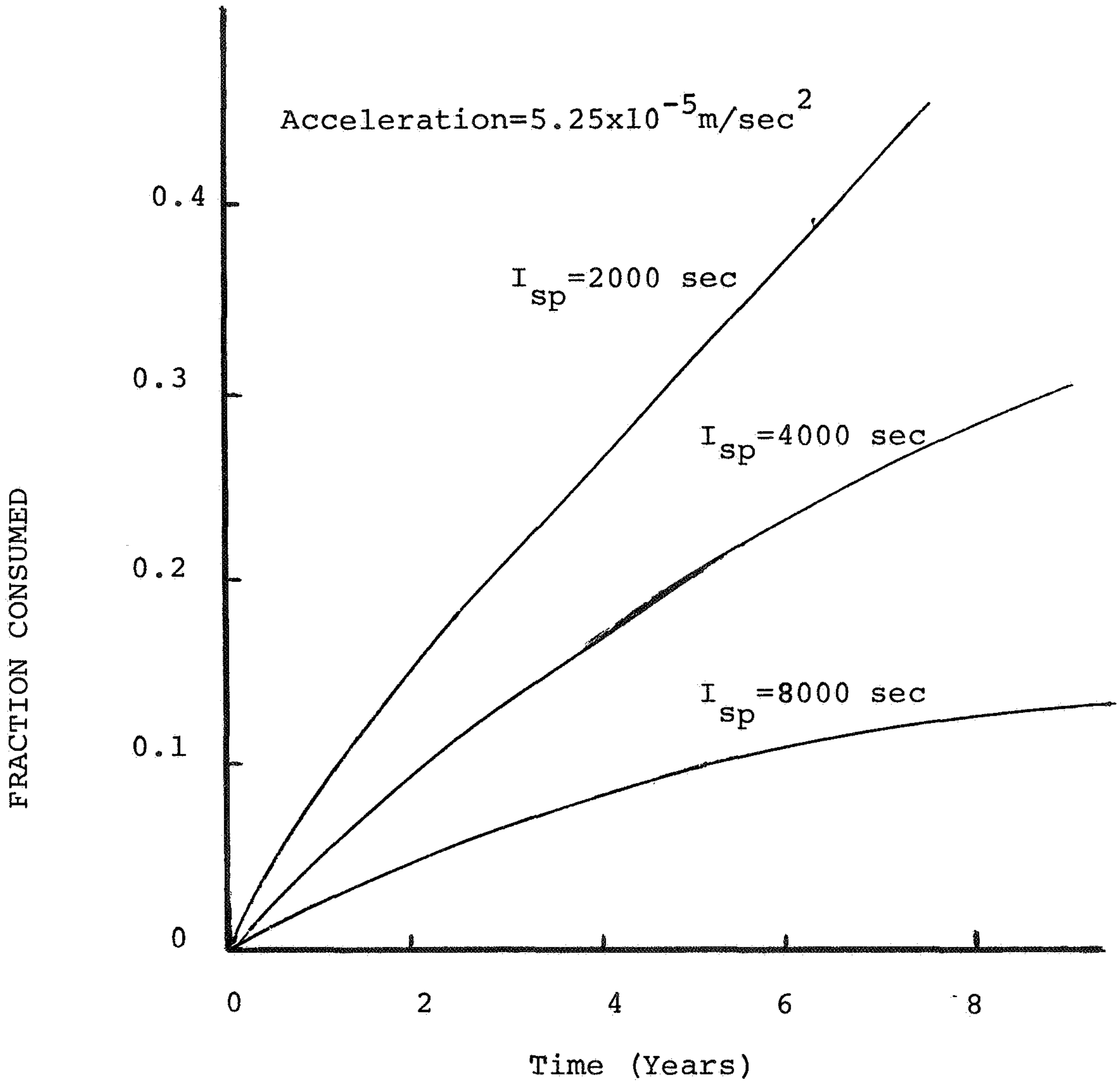

FIGURE 19.2-6 FRACTION OF FUEL CONSUMED vs TIME ALONG THE ORBITAL AXIS FOR HUMMINGBIRD SATELLITE

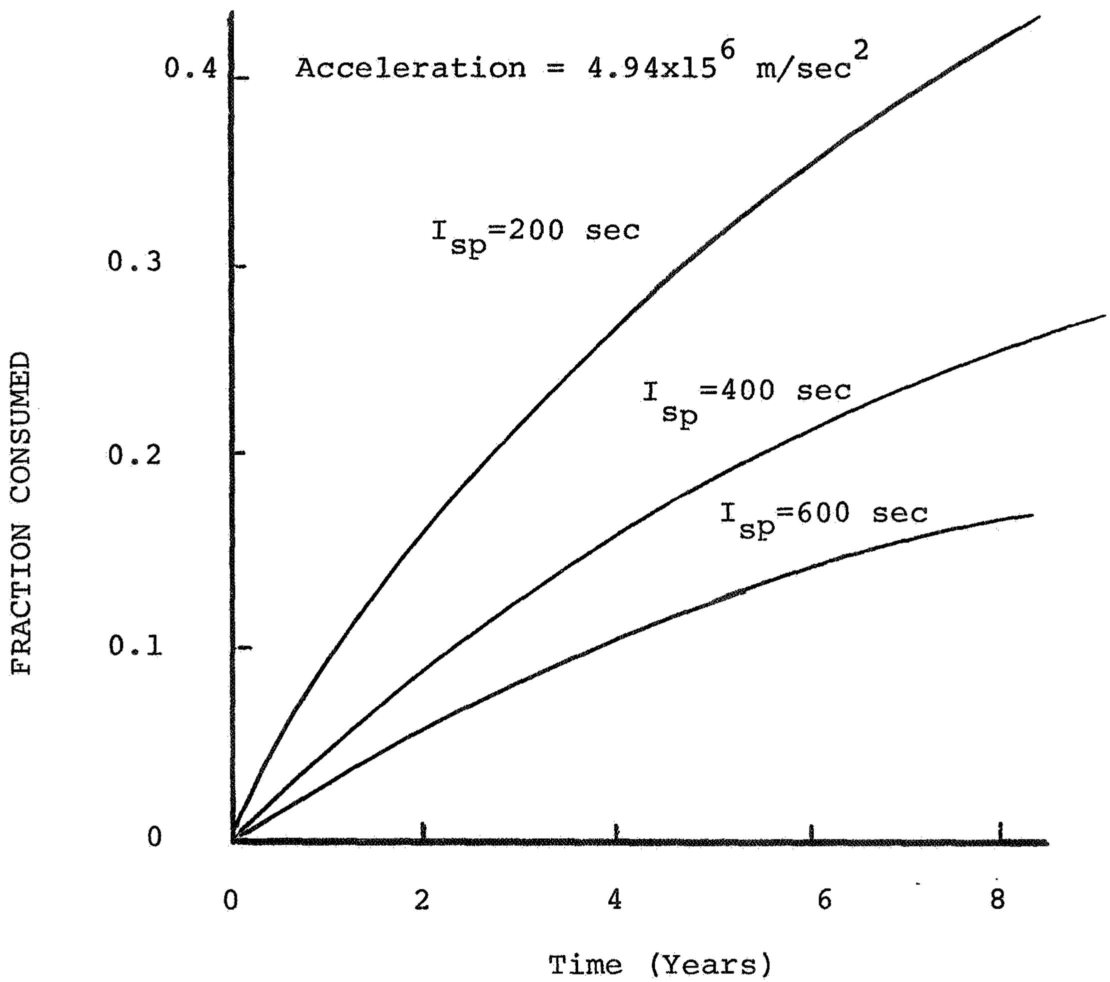

FIGURE 19.2-7 FRACTION OF FUEL CONSUMED VERSUS TIME FOR HALO RELAY SATELLITE

The acceleration which the Hummingbird must overcome as calculated earlier, is $5.25 \times 10^{-5} m/sec^2$ for an altitude of 3300 km along the orbital axis. Assuming an average weight of 1100 pounds to be maintained in orbit, it would require the thrust level for the propulsion system to be

$$F = \frac{1000}{32.2} \ (1.72 \times 10^{-4}) = 5.88 \times 10^{-3} lb.$$

Over a three year period, a total impulse of 550,000 lb-sec. would be required. The associated velocity impulse for this life time is

$$\Delta v = I_t \ (\frac{g}{m}) = 5500{,}000 \ (\frac{32.2}{1100}) = 4{,}900 \ m/sec^2$$

Thus, an ion engine, a low thrust high specific impulse system, should be considered for orbital maintenance if a Hummingbird relay satellite is to be deployed.

19.2.2.2 Halo

The average acceleration the halo satellite relay will be subjected to is 4.94×10^{-6} m/sec^2. This assumes a near circular orbit about the L_2 libration point with radius 3300 km in length. Figure 19.2-7 illustrates the percentage fuel consumed by the Halo as a function of time for various values of specific impulse. The specific impulse considered here is an order of magnitude less than that for the Hummingbird, since the acceleration which must be overcome is on a much lower level. Unlike the Hummingbird, the Halo relay is required to perform thrust maneuvers once each half period to maintain itself in a circular orbit. This requires imparting a velocity impulse of 2.02 m/sec each half orbit which is approximately 7.5 days. Assuming a spacecraft life time of three years, the total Δv required by the halo to keep on station is 297 m/sec. This is substantially less than that required by Hummingbird for orbital maintenance over the same period of time.

19.2.3 Polar Satellites

It is more difficult to talk about station keeping for polar satellites than for those which are placed about the L_2 libration point. This is because there are so many parameters which must be considered if there is to be an accurate analysis performed. Unlike the libration satellite, where a value can readily be obtained giving the Δv requirements for station keeping, the lunar relay requires more detailed specifications first. For example: What is the minimum acceptable deviation allowed for the orbital elements; how many satellites should be deployed in each orbit; at what altitude should the satellite revolve; how many trim maneuvers should be performed during a given period of time; what is the given period of time; what is the minimum acceptable overlap for continuous communications coverage; and what is the desired lifetime of the relay satellite? Answers to these and possibly other questions should be specified if one is to optimally compute the orbital maintenance of a lunar satellite.

Ideally, a satellite placed in a circular orbit would remain there indefinitely, if the moon were perfectly spherical in shape and could be considered as an isolated body free from gravitational effects of other bodies. However, due to the perturbations of the moon, earth or sun, the orbits and phase angle between satellites will deviate from their nominal values. Unless this variation is kept under control by occasional trim maneuvers, the satellite eventually will find itself in a non-useful position. A direct consequence of these third body effects is to change the orbital characteristics from circular to elliptical in nature i.e. assuming the satellite is initially inserted into a perfect circular orbit (eccentricity = 0, ideal case) the perturbations due to the moon, earth and sun cause an eccentricity to develop and grow. For a particular variation in eccentricity, the polar satellite located at the higher altitude will achieve this value in a shorter period of time. This is illustrated in Table 19.2-4.

As stated earlier, if certain quantities are specified, it is possible to estimate the orbital maintenance cost (Δv) for polar satellites deployed in the same orbit at an altitude of 6000 nautical miles. In addition assume that the acceptable deviation in phase for each satellite is 20^{o}. Then the average velocity impulse required by each satellite is 64.5 m/sec every 200 days. For a desired lifetime of three years, each polar satellite will require a total Δv of 386 m/sec. It should be pointed out that value does not include; a) any initial plane changes required by the satellite to adjust into circular orbit. b) the phasing requirements between other sets of satellites in different polar orbits. In addition, it should be noted that if the frequency of the trim maneuvers were increased, it is

TABLE 19.2-4 ECCENTRICITY EFFECT OF THREE BODY SYSTEM

Altitude (n.m)	Time (yrs)
2,000	1.15
4,000	0.52
6,000	0.33
8,000	0.21
10,000	0.15

*Table prepared for a variation in eccentricity from 0.001 to 0.01

possible to reduce the Δv requirements even further. This frequency can be optimized and is left for further study.

19.2.2.4 Conclusion

The results of this section are summarized in Table 19.2-5.

Table 19.2-5 TOTAL DEPLOYMENT COST

Satellite System	Orbit Injection (M/Sec)	Orbit* Maintenance (M/Sec)	Total
Hummingbird	376	4,900	5,276
Halo	376	297	663
Lunar Polar Orbit	780	386	1,166

*3 Year Life

*Three Satellites

Obviously, on a deployment cost basis, the Halo appears very attractive. It should be noted the polar system requires a minimum of nine satellites for full and continuous coverage of the moon. There are still two major points to consider, one of which involves the deployment of a satellite or any other craft in the vicinity of the L_2 libration point. This is still an unproven concept that involves an element of risk which should be considered. On the other hand, a single vehicle has been made to orbit the moon successfully and the techniques and technological knowhow are a matter of record. However, if the decision is made to choose the polar system. this still involves the risk of placing three or more satellites simultaneously into lunar orbit.

19.3 Launch Vehicle Considerations

The cost of space missions is largely determined by the launch vehicle chosen to accelerate and elevate the spacecraft to their mission velocities and altitudes. Since boosters are presently non-recoverable, the transportation costs in delivering a payload to orbit are relatively high. The objective of this analysis is to select possible candidate launch vehicles and to compare the replacement costs. In order to select a launch vehicle, the payload it is required to deliver must be totally specified. Table 19.3-1 lists the expected weight for each satellite.

TABLE 19.3-1 RELAY SATELLITE WEIGHT

Satellite	Weight (lbs)
Halo	1200
Hummingbird	1300
Polar	2000*

*This is a maximum value

A comparison on a cost basis can readily be made by utilizing these values in conjunction with Table 19.3-2.

Obviously either libration satellite can be independently launched by a vehicle with the capability of a Tat-Delta with the hydrogen-oxygen second stage or greater. The polar system of satellites, however, requires the launching of several satellites simultaneously into lunar orbit so that a smaller cost would be incurred. The Titan 3C, which has successfully

TABLE 19.3-2 LAUNCH VEHICLE CAPABILITIES SUMMARY

Launch Vehicle	Escape Payload * (lbs)	Replacement Costs ($M)
TAT-Delta-3 Castors - FW4	380	3.01
TAT-Delta-3 Castors - TE 364	470	3.07
SLV3A-Burner 2	625	4.3
TAT-Delta-6 Castors - TE 364	710	3.36
TAT-Delta-9 Castors - TE 364	830	3.62
TAT-Delta-3 Castors - Hoss - TE 364	1150	3.76
TAT-Delta-6 Castors - Hoss	1280	4.05
TAT-Delta-9 Castors - Hoss	1480	4.31
Titan 3X-Agena	2300	8.6
SLV3C-Centaur	2800	10.8
SLV3X-Centaur	4800	10.8
Titan 3C	5000	17.2
Titan 3D-Centaur	12500	16.8

19.4 Communication Coverage of The Moon

Of prime importance in establishing a relay satellite system is to provide adequate communications coverage. Initially this might mean to support existing Apollo programs with partial communications between the farside of the moon and the earth. However, the future may necessitate communications between any two points on the surface of the moon. This is evident in the design of the MULE, which must be capable of unmanned exploration of the lunar surface. The control site for MULE is probably located at the Space Base.

Figure 19.4-1 points out the relation between altitude and percent of lunar hemisphere covered by a satellite, e.g. the Halo or Hummingbird. A satellite stationed about the L_2 libration point illuminates approximately 97.3% of a lunar hemisphere at any given time assuming the communications network included the earth on one side and a libration satellite at the backside, then only a small strip ranging between 30 and 80 nautical miles wide would remain uncovered. A single polar satellite, although not located as far as the libration satellites, will have its coverage further restricted by the additional constraint that the earth and surface terminal be viewed simultaneously. This places a burden on the antennas, requiring them to cope with wide angular displacements. Relay between polar satellites is totatly unfeasible due to the severe requirements imposed on the communications system within the satellite. A minimum of nine satellites equally phased in polar orbits 60^{o} apart will provide

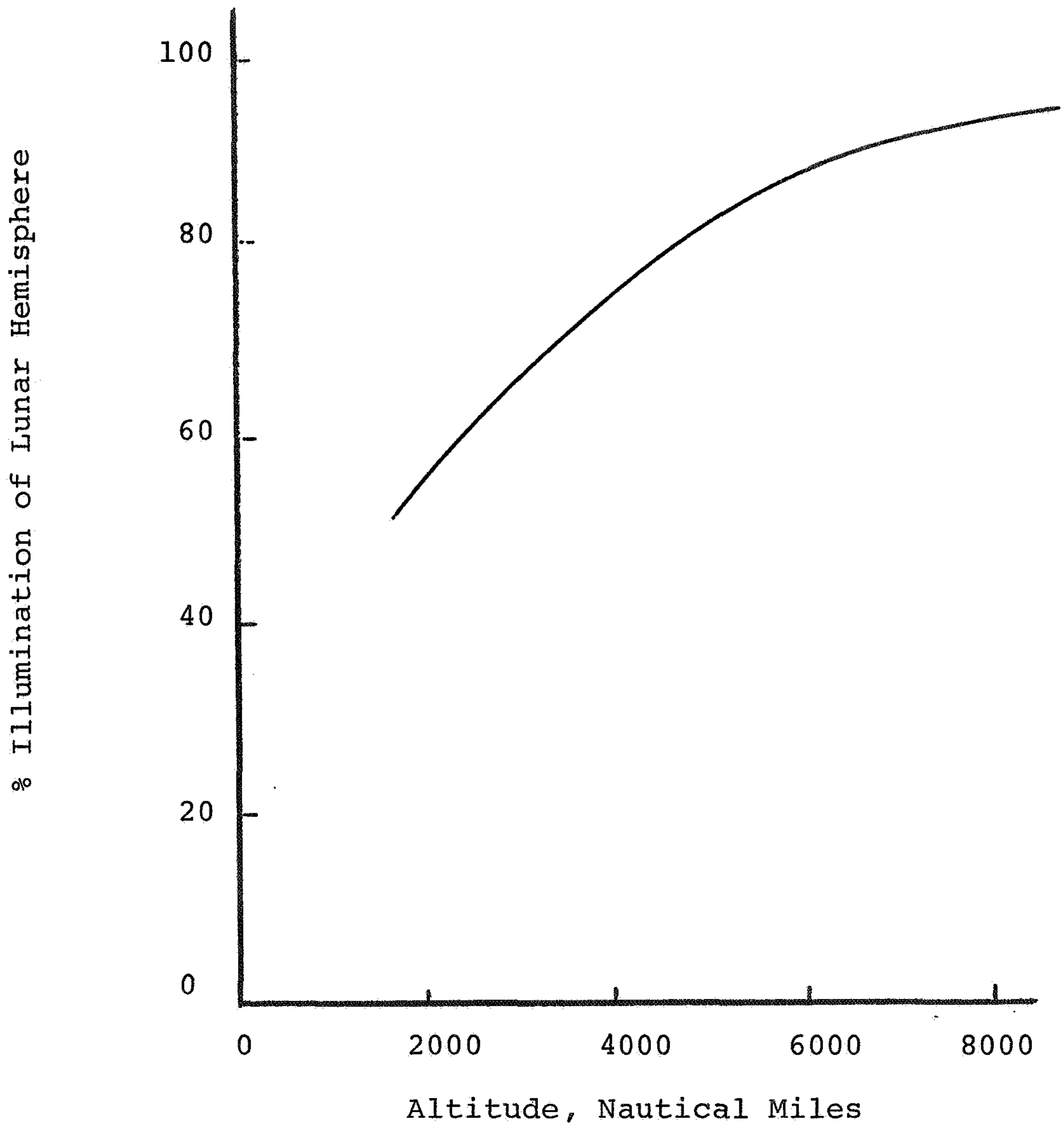

FIGURE 19.4-1 PERCENT ILLUMINATION OF LUNAR HEMISPHERE vs ALTITUDE

continuous coverage of the moon. The amount of overlap will depend on the altitude employed. Another disadvantage is the difficulty in tracking and acquisitioning of low altitude polar satellites from the lunar surface. For example, it would be foolish to have an astronaut waste his time to acquire the next polar satellite just before the present relay has achieved a useless state. For low altitudes, a useful state might only amount to a few hours. On the other hand, the libration satellite presents a stationary target and may be acquisitioned fairly easily through directing a high gain antenna according to a tone modulated carrier emanating from the relay. However, the price that is paid for this convenience is a significant increase in path loss. This has been computed to be an increase of 1.4 DB path loss for an increase in range separation from 3.84×10^{-5} km to 4.8×10^{-5} km.

19.5 Continuous - Motion Remote Control Driving

The lunar mobility aid places an additional constraint on the choice of communications system employed. Since the lunar roving vehicle is expected to make long unmanned traverses along the lunar surface, continuous communications are required to operate and monitor the system. In addition, the time delay accrued during transmission between the mobility aid and control site determines the speed of the vehicle. If the delay becomes too large the vehicle will not be able to successfully complete missions of long duration. Enumerated in Table 19.5-1 are various locations for controlling an unmanned lunar vehicle

TABLE 19.5-1 REMOTE CONTROL TRANSMISSION TIME DELAY

Control Site	Scheme Employed	Transmission Delay (sec)
Earth	Vehicle located on near side of moon	1.39
Earth	Communication satellites at far side libration point	1.83
*LOSS	Continuous control for short periods of time Ex. 800 mile orbit ≈ 55 min 60 mile orbit ≈ 10 min	0.10
*LOSS	Communication satellites at both near and far side libration points	0.92
*LOSS	**Polar satellite relay system	0.12
Space base on lunar surface	Vehicle on same side of moon as space base **Polar relay satellite system	0.10
Space base on lunar surface	Vehicle on side where space base located, one communication satellite at libration point, also on same side	0.48
Space base on lunar surface	Vehicle on opposite side of moon from space base, communication satellites located at both libration points	0.92
Space base on lunar surface	Vehicle on opposite side of moon from space base **Polar relay satellite system	0.12

*Assumed LOSS in 60 miles polar orbit above lunar surface

**Assumed polar orbits of 60 miles above lunar surface for each polar communication satellite.

for the purpose of comparing the communications coverage given by each satellite system. Depending on the site and scheme utilized different transmission delays are encountered. It should be noted these are "maximum" one way transmission delays which include the electronic processing time. The vehicle may be located anywhere on lunar surface unless specifically stated otherwise. As an example, if the largest acceptable delay is 2 seconds for continuous motion, remote driving control must be initiated at a site other than the earth. At first glance, it appears that the polar satellite system will provide the shortest transmission delay and thus allow the vehicle to accomplish the more stringent missions. This is an area which requires further study.

19.6 Conclusions and Recommendations

19.6.1 Conclusions

Polar Satellite Disadvantages:

1. Each communication satellite is occulted by the moon during each orbital period. The altitude determines the length of time the communications will be blacked out.
2. A minimum of nine satellites are required for complete coverage of the moon.
3. The launch vehicle employed must be capable of delivering a heavier payload.
4. Deployment cost is considerably higher than a libration satellite (780 m/sec vs 376 m/sec).

5. Station keeping is lightly greater than the Halo satellite (386 m/sec vs 297 m/sec).
6. A complex system is required for orbital stability and phase control.
7. Tracking low orbiting relay satellites by surface terminals is difficult.
8. Continuous shift from satellite to satellite is necessary.
9. The risk involved with deploying more than one satellite into lunar orbit at the same time is grater.

Libration Satellites Disadvantages

1. Path loss increased by 1.4 DB.
2. The transmission delay incurred during unmanned operations of the lunar roving vehicle may be critical.
3. The libration satellite system is an unproven system and a large risk factor should be considered.

19.6.2 Recommendations

The analysis performed here, although limited in scope, strongly emphasize the need for some type of intermediate relay communications network about the moon.

The author, based on overwhelming evidence presented in this report, recommends a Halo libration satellite be placed about the L_2 stagnation point. It is believed that such a satellite will support and fullfill the immediate goals of the space program. However, keeping in mind the objectives of this study, it is further suggested that at some later date a second satellite be placed exactly at the L_1 libration point and thus, the Halo situated at the other libration point would be required to be maneuvered into a slightly larger orbit so that it maintains an unobstructed view of the L_1 libration point. Figure 19.6-1 illustrates the proposed libration satellite system. It is hoped that this report will provide the basis for further study.

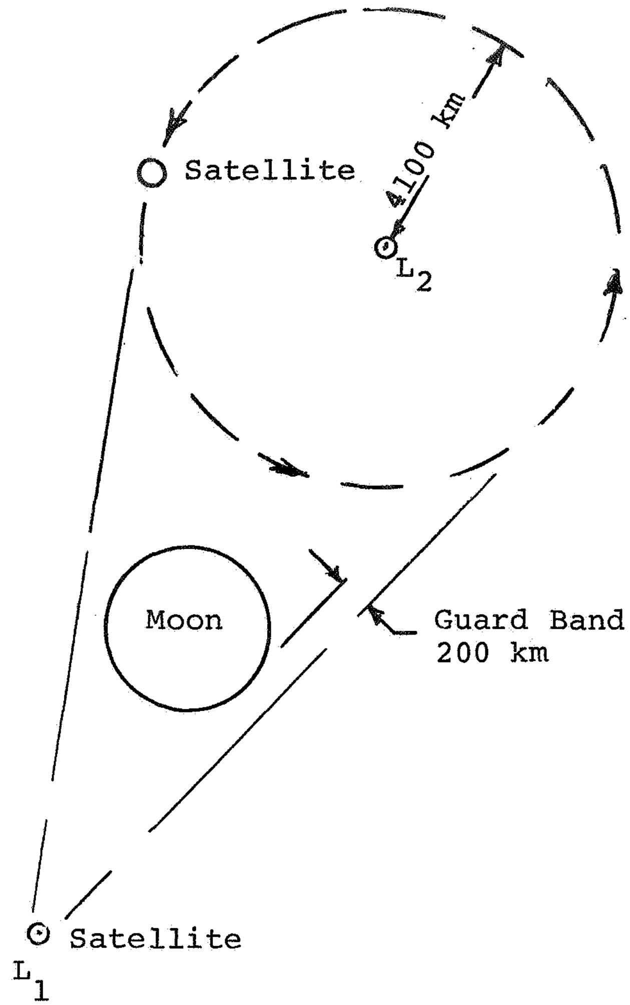

FIGURE 19.6-1 PROPOSED LIBRATION SYSTEM OF SATELLITES

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CHAPTER 20

SYSTEM COMPARATIVE COST ESTIMATIONS

J. E. Sneckenberger

An integral part of the methodology developed in Chapter 9 for evaluation of alternative mobility systems was the need for a reliable procedure for arriving at cost estimates of the first unit cost and the Research Development, Test, and Evaluation cost for any proposed candidate mobility system. During the conception of the evaluation methodology, no alternative to the use of cost estimates for first unit and RDT&E was found. Thus, although costing is difficult and uncertain, a procedure for calculating the first unit and RDT&E cost of a mobility system became an essential element of the effectiveness cost methodology.

Costing the first unit and RDT&E costs for mobility systems envisioned for ten years and longer into the future constitutes costing hardware involving new materials, manufacturing processes, etc. The resulting uncertainty in cost estimates reemphasizes the importance of comparing estimates rather than stressing the absolute value or alternative systems costs.

20.1 Method of Costing Employed

A well-conceived cost estimating procedure for advanced space systems comprises the identification of specific subsystem items, available historical data, and well developed cost estimating relationships (CER'S). The calculation of the cost estimate is accomplished through the use of the CER's associated with a specific cost category parameter to estimate subsystem hardware costs which can be summed to obtain the system level cost. It is important to insure that the costing method developed be made consistent for all candidate systems.

In order to permit the committee on subsystem description flexibility within its task of defining a set of subsystems which adequately covers the spectrum of past, present, and projected lunar mobility systems, the costing method conceived was developed around cost categories rather than the, as then, undefined subsystems. Cost categories are descriptions of the general system oriented for cost purposes. They provide a convenient technique to assist in establishing the cost of an arbitrary condidate mobility system consisting of various subsystems in terms of the available historical costing data. That is, the cost categories should highlight the key features of they system while, at the same time, permitting maximum use of cost data collected from past systems.

Subsystem specifications involving systems which are basically in the conceptual design stage generally are not defined with exactness. A survey of available contractor reports of lunar surface transportation concepts was performed to provide a list of the presently identified subsystem descriptions. A summary of the various hardware components defined is shown in Table 20.1-1. These hardware components were then grouped into the cost categories shown in Table 20.1-2. The cost categoreis chosen were selected in part to correspond with cost data of available space vehicle concepts. Word descriptions for each of the cost categories are as follows:

Structure:

The structure category consists of the main vehicle chassis, including cabin module with all supporting frames, platforms, attachments (cranes, manipulators), compartments, etc. The cabin module, which provides areas for working, sleeping, experimentation, etc, is comprised of bulkheads, flooring, windows, facilities for egress/ingress, and environmental shielding (heat, radiation, etc.). Also included in the structure category are support structures (hatches, tanks, tubing, etc. for crew equipment, instrumentation, fuel, payload, etc., as well as provisions for initial delivery, lunar sample storage and, for adapting the vehicle to assist in mission operations (core sample drilling, nuclear power generator repair, construction, personnel transportation).

Locomotion:

The locomotion category consists of hardware which provides suspension and controlled translation of the vehicle. In addition to the engines (electric motors), the locomotion category consists of hardware items such as the wheels tracks, idler, steering and braking controllers, and suspension assemblies. It also includes such items as emergency braking devices.

Crew Station:

The crew station category comprises all vehicle hardware which provides for the comfort, well-being, support, and safety of the crew. It includes such hardware items as couches, restraints, displays (navigation, locomotion, thermal control, electrical power, etc.), crew provisions (personal hygiene equipment, feeding and waste disposal items, etc.), as well as items which maintain and monitor the physical and chemical condition of the atmosphere within the vehicle.

Astrionics:

The astrionics category is composed of vehicle hardware which contributes directly to the functions of sensing, referencing, computing, commanding, and receiving/transmitting. It includes all items of equipment related to navigation, guidance, stabilization, hazard detection, remote control, telemetry, and communications such as cameras, directional gyros, accelerometers, radar, remote sensors, transmitters, decorders, and antennas (Omni, S-Band).

Electrical Power:

The electrical power category includes all vehicle hardware which generates, converts, controls, and distributes electrical power within the mobility vehicle. Typical items of electrical power equipment for a mobility vehicle are batteries, fuel cells, radiators, solar arrays, RTG's, conditioning units (regulators, inverters, chargers) and distribution devices. It also includes such items as cabling and adapters for electrical compatibility between vehicle subsystems.

These cost categories are thus broad enough to encompass the present state of the art of lunar surface transportation concepts.

TABLE 20.1-1. SUMMARY OF MOBILITY SYSTEM HARDWARE COMPONENTS

Airlock	Mobility
Astrionics	Navigation
Chassis	Navigation and Control
Command and Control	Navigation and Guidance
Communications	Orientation Controls
Crew Station	Plenum
Crew Systems	Power
Cryogenic Storage	Propulsion Unit
Electrical Power	Remote Control
Electronics	Space Support Equipment
Environmental Control	Structure
Hazard Detection	Suspension Assembly
Instrumentation	Thermal Control
Landing Gear	Thermal Protection
Life Support	Wheels

Cost data, eg. RDT&E cost per pound, is available for a number of the above hardware components.

Note: Fuel, payload, etc. are not considered because only the mobility system dry weight is involved.

TABLE 20.1-2. MOBILITY SYSTEM COST CATEGORIES

Cost Category	Hardware Components
Structure	Airlock, Structure, Chassis, Cryogenic Storage, Thermal Protection, Plenum, Propulsion Unit
Locomotion*	Landing Gear, Mobility, Suspension Assembly, Wheels
Crew Station	Crew Station, Crew Systems, Environmental Control, Life Support, Space Support Equipment, Thermal Control
Astrionics	Astrionics, Command and Control, Communications, Electronics, Hazard Detection, Instrumentation, Navigation, Navigation and Control, Navigation and Guidance, Orientation Controls, Remote Control
Electrical Power	Electrical Power, Power

* "Mobility" is preferred; however, the obvious dual usage of mobility to describe a subsystem as well as the main system is undesirable.

20.2 Development of Cost Estimating Charts

The main difficulty in the development of meaningful cost estimating relationships arises because of uncertainty in precise definition or subsystem items and their associated cost. The cost data, which includes actual system data, estimates made from current programs, and estimates based on data from other similar studies, should be analyzed statistically and presented in analytical form for each cost category. However, since there was insignificant data available to do statistical studies leading to analytical CER's, graphical methods offered another approach to the development of cost estimating relations. Further, for this study which in Phase II considered only near-ground concepts, emphasis was placed on the development of graphical cost estimating relationships for the rover class vehicle.

Figures 20.2-1 through 20.2-5 for first unit and figures 20.2-6 through 20.2-10 for RDT & E costs were generated from consideration of the available data updated to 1970 dollars, previous similar graphical cost estimating relationships, etc. As shown, weight was considered as the independent parameter for each of the cost categories. Lines of constant complexity of hardware concepts provide bounds for estimating the value of the cost/pound of each hardware category.

20.3 Cost Calculations for Synthesized Vehicle

After a proposed mobility system has been described to the subsystem level, it is then necessary to allocate the weights of subsystems to the appropriate cost categories. Table 20.3-1 has been prepared using the weights of the various system components identified in connection with the weight breakdown of the synthesized mobility system. The distribution of component weights between cost categories requires as complete a subsystem description as possible.

An estimate of the cost/pound for first unit and RDT & E costs was determined for each category using the appropriate graphical

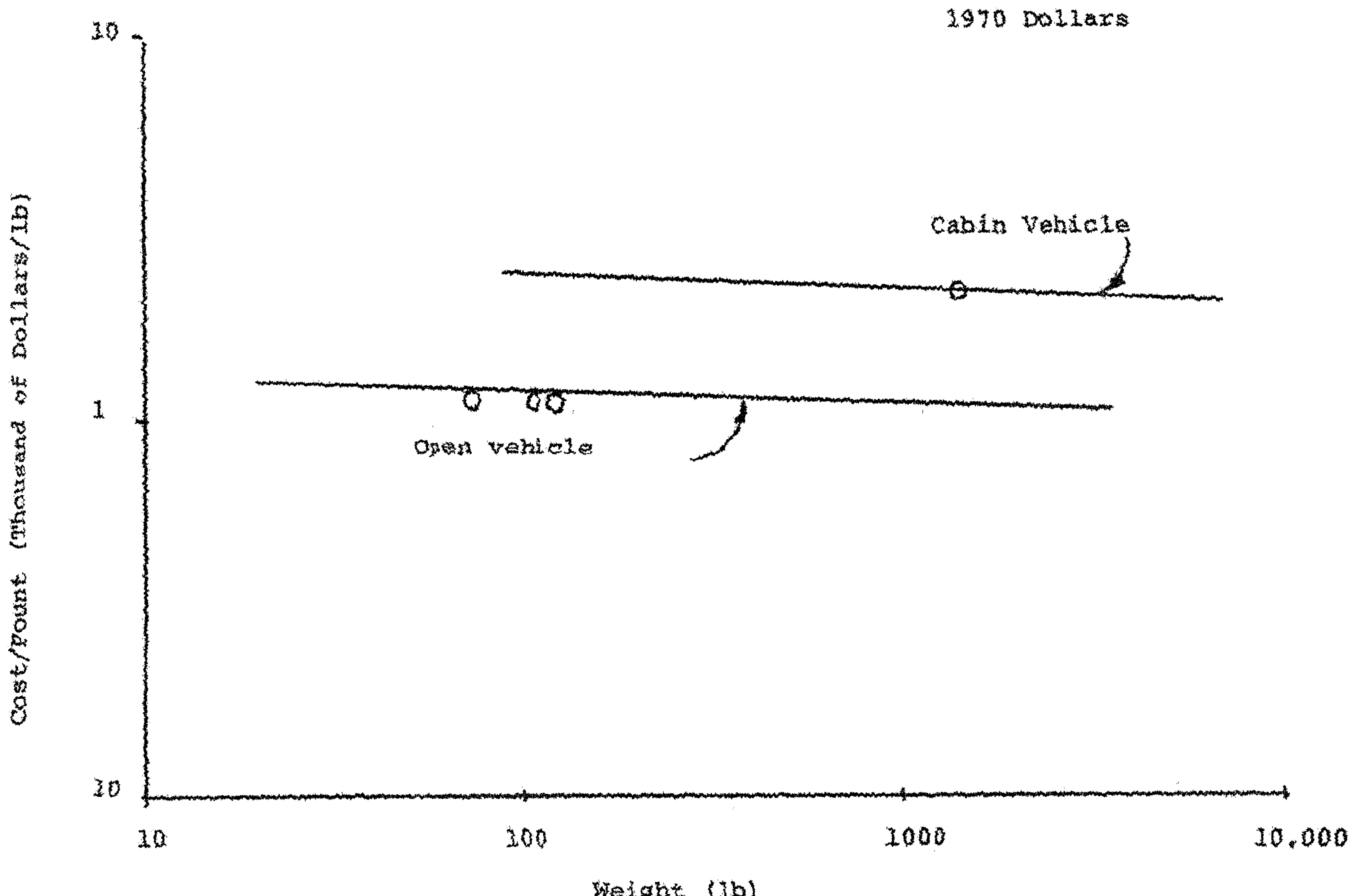

FIGURE 20.2-1 STRUCTURE FIRST UNIT COST, ROVER CLASS VEHICLE

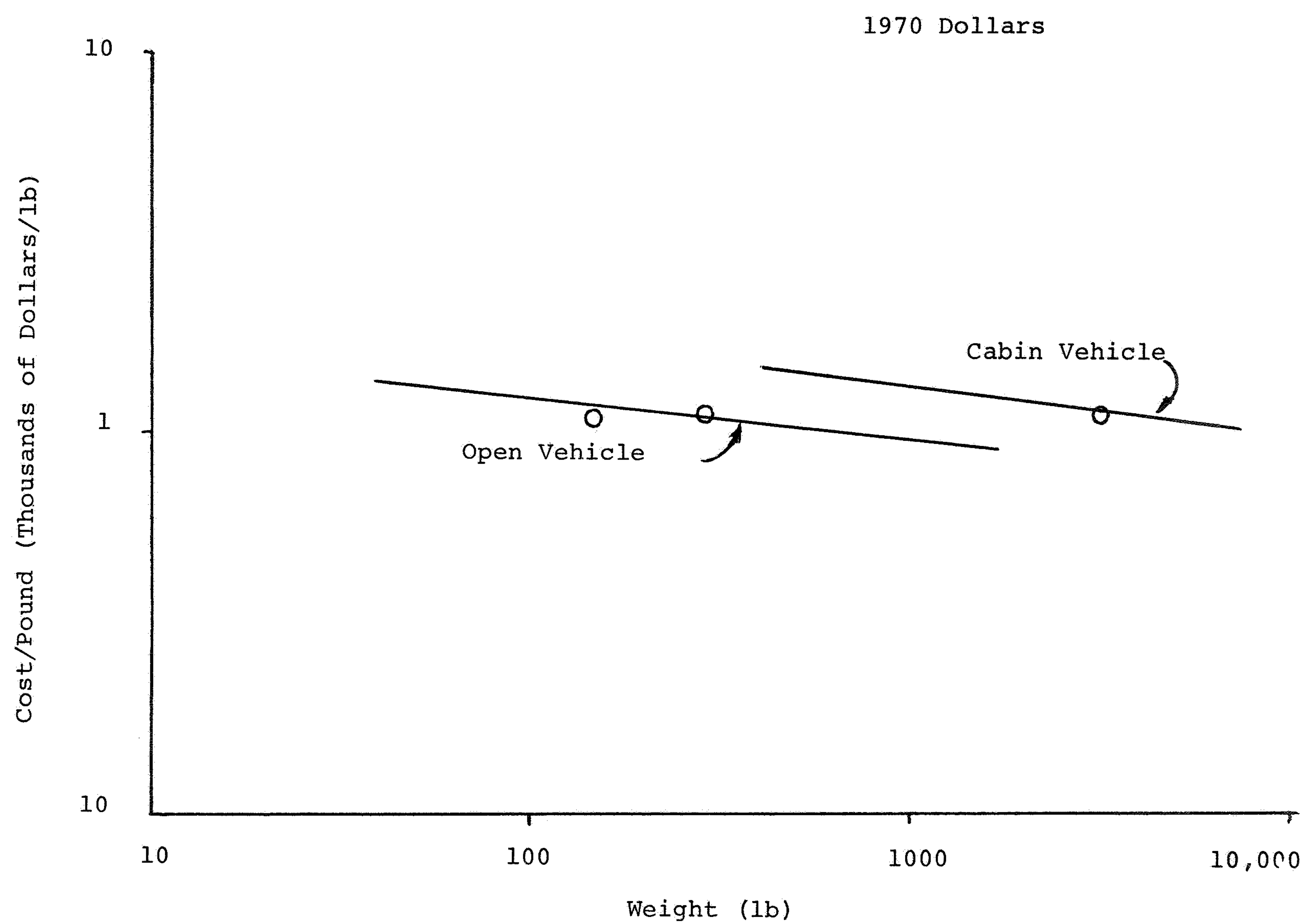

FIGURE 20.2-2 LOCOMOTION FIRST UNIT COST, ROVER CLASS VEHICLE

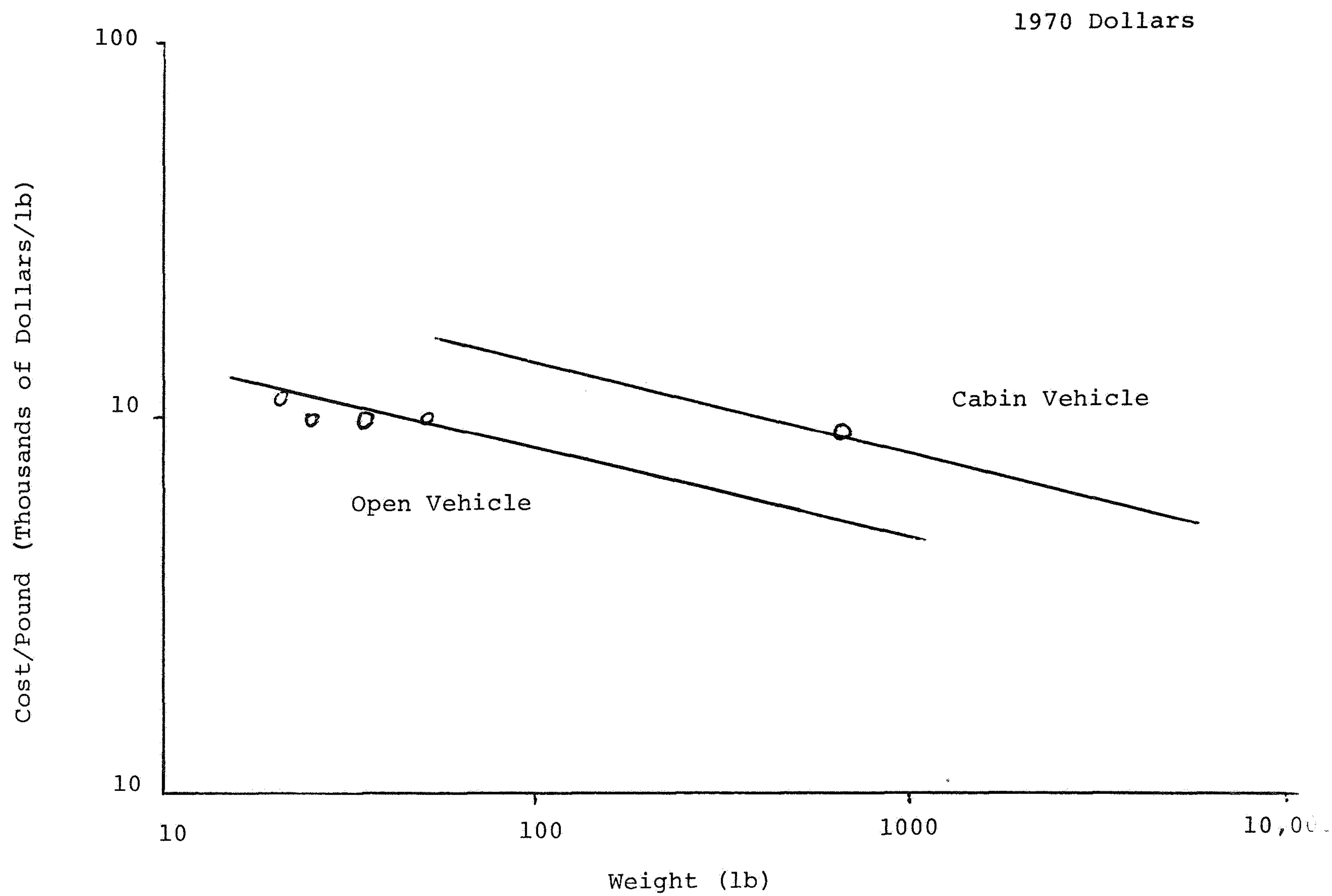

FIGURE 20.2-3 CREW STATION FIRST UNIT COST, ROVER CLASS VEHICLE

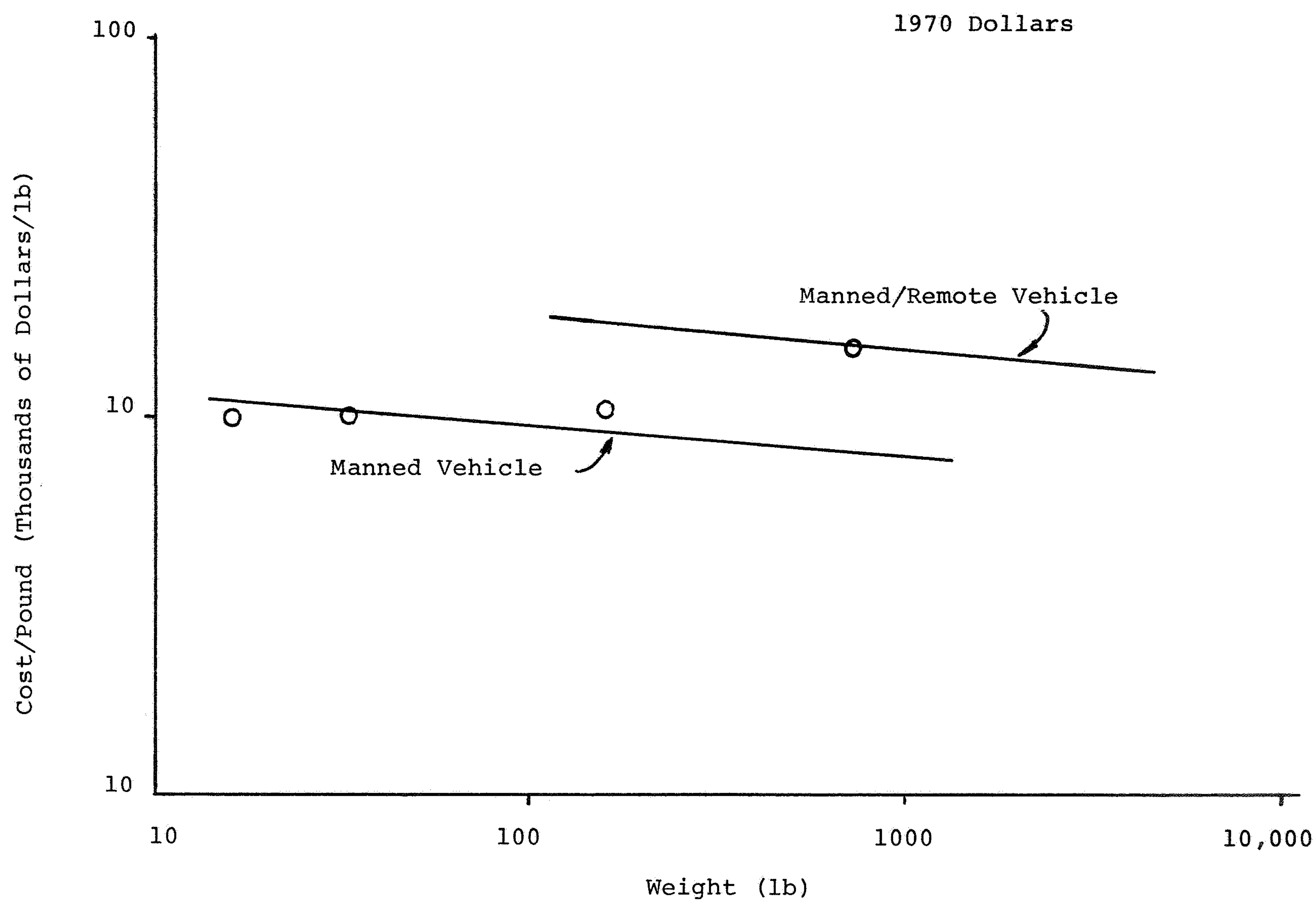

FIGURE 20.2-4 ASTRIONICS FIRST UNIT COST, ROVER CLASS VEHICLES

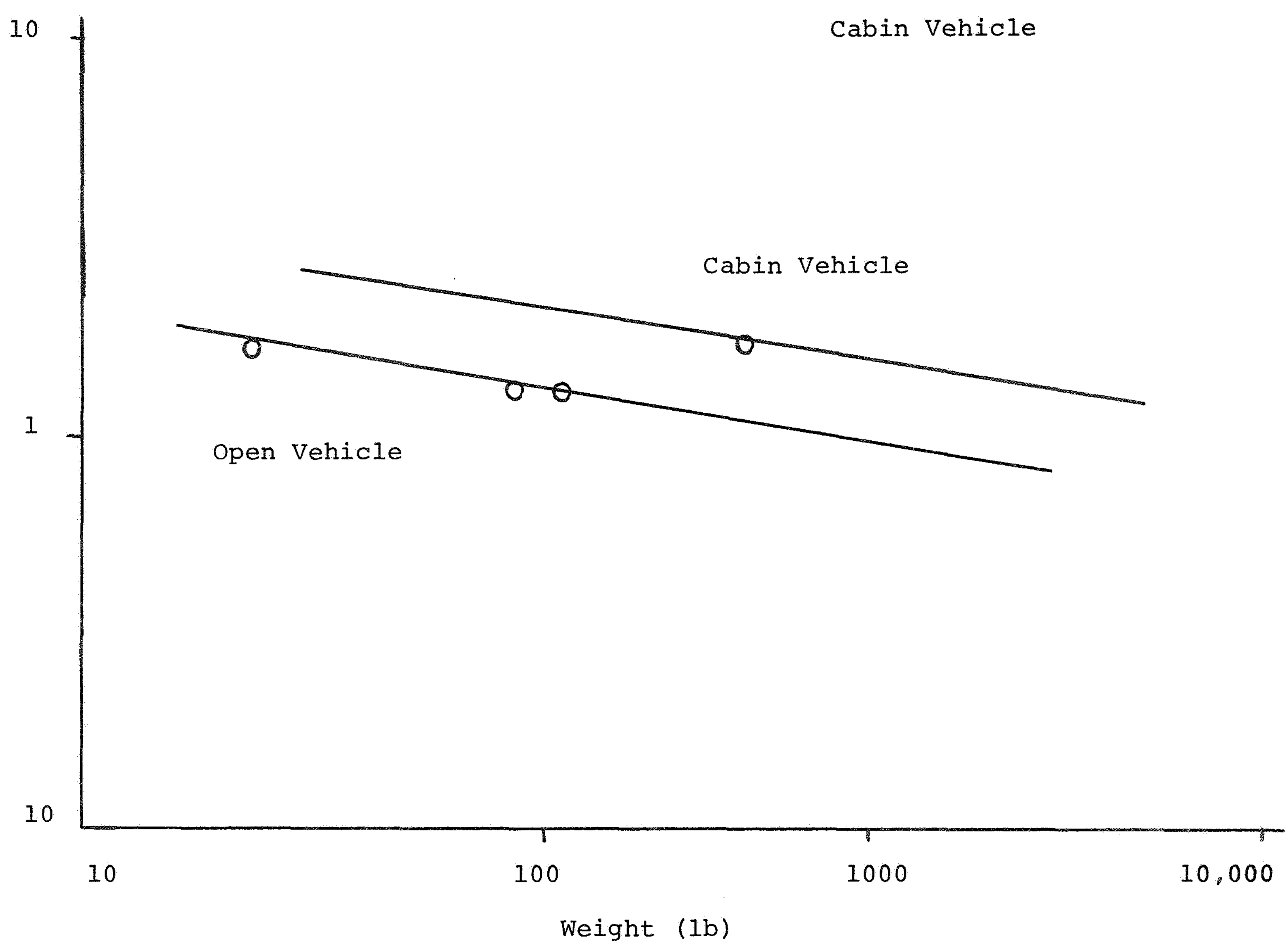

FIGURE 20.2-5 ELECTRICAL POWER FIRST UNIT COST, ROVER CLASS VEHICLE

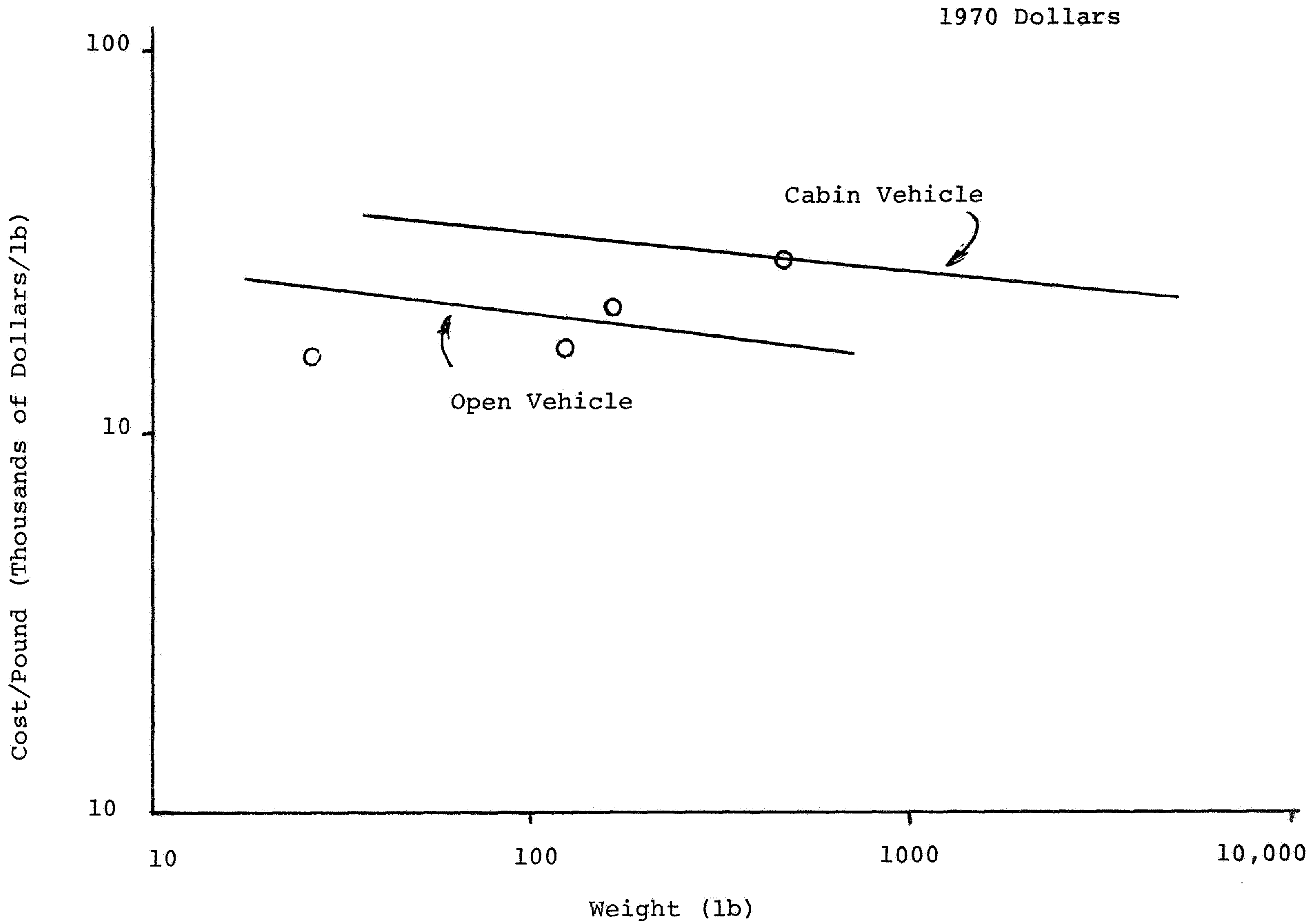

FIGURE 20.-6 STRUCTURE RDT&E COST, ROVER CLASS VEHICLE

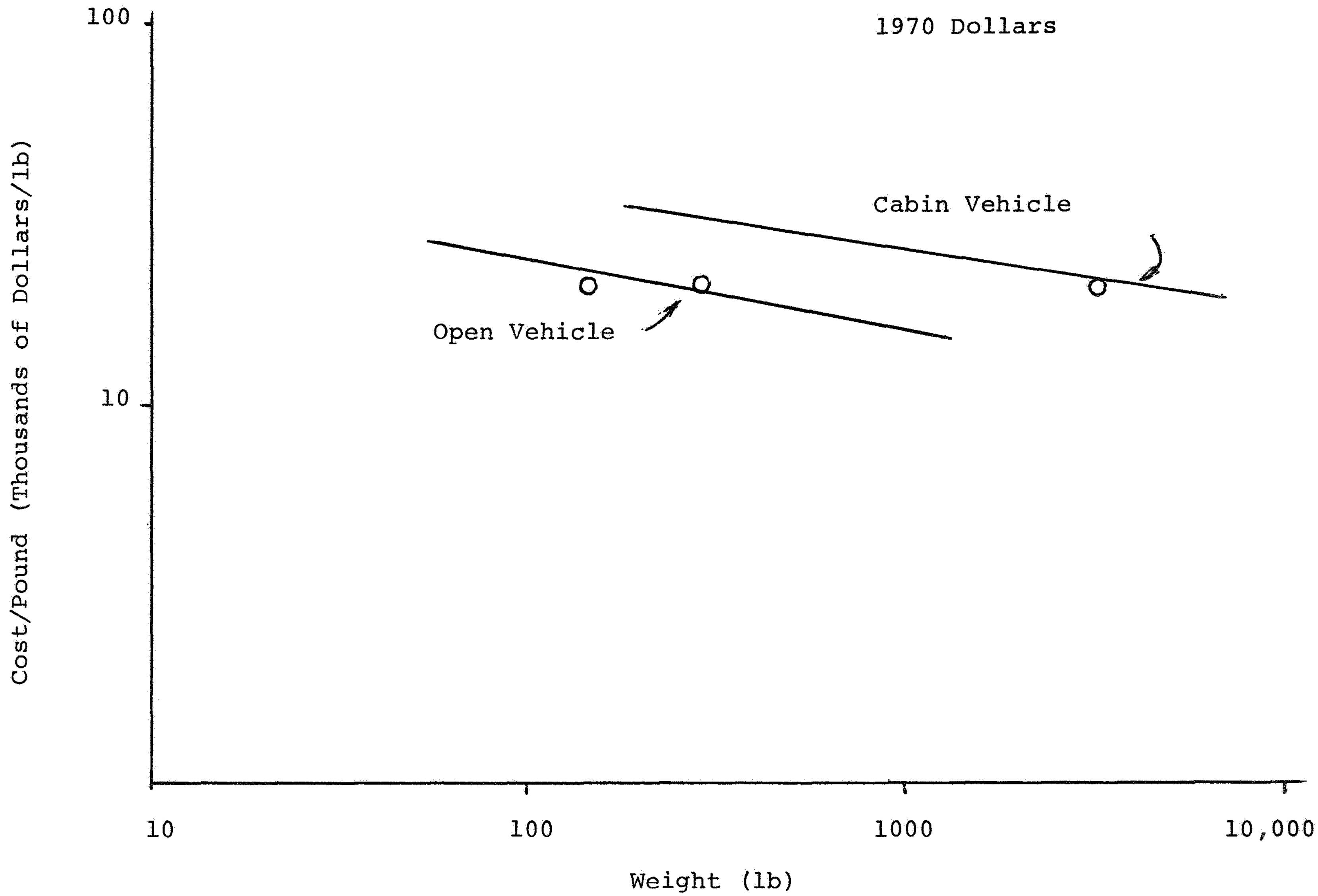

FIGURE 20.2-7 LOCOMOTION RDT&E COST, ROVER CLASS VEHICLES

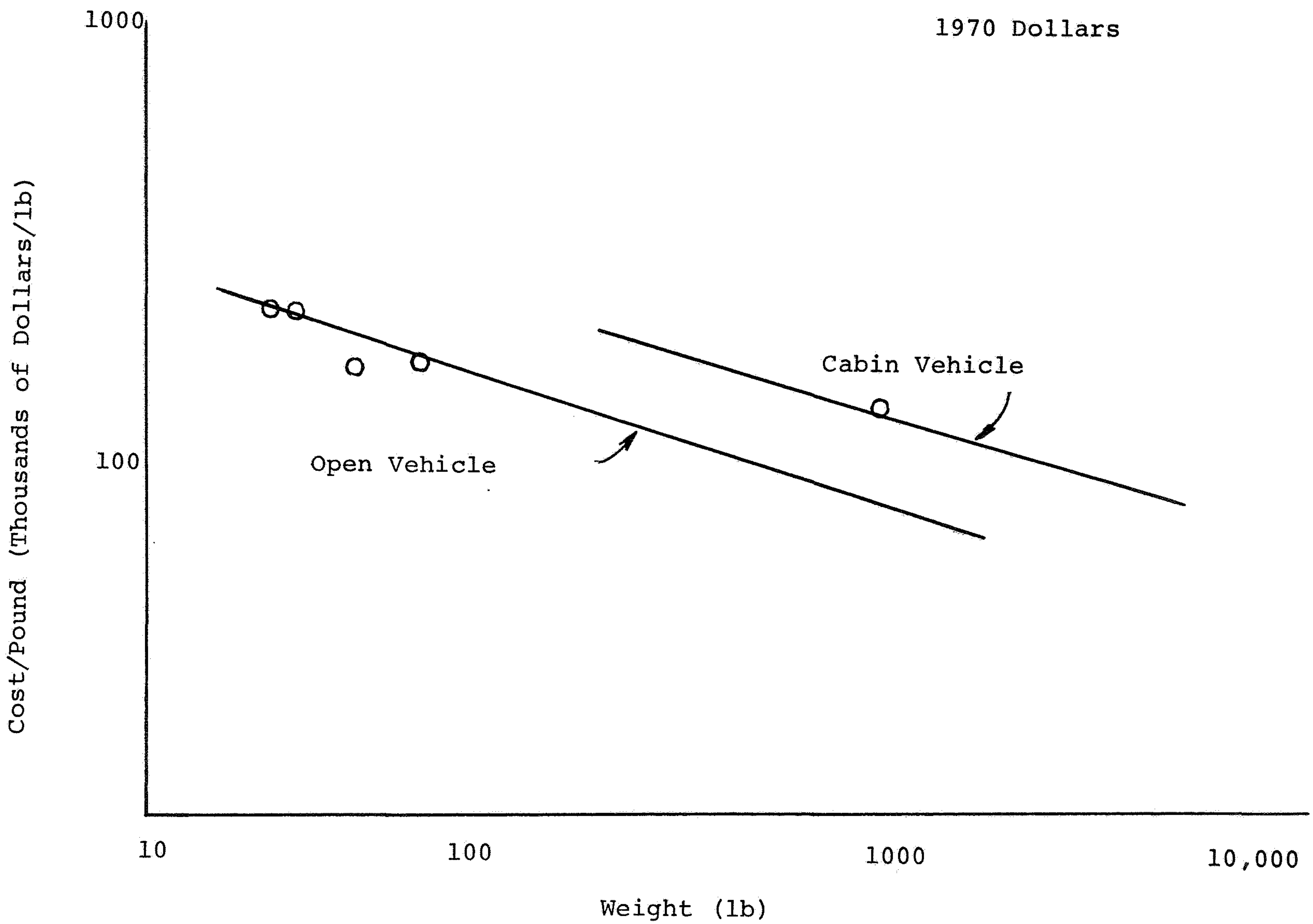

FIGURE 20.2-8 CREW STATION RDT&E COST, ROVER CLASS VEHICLE

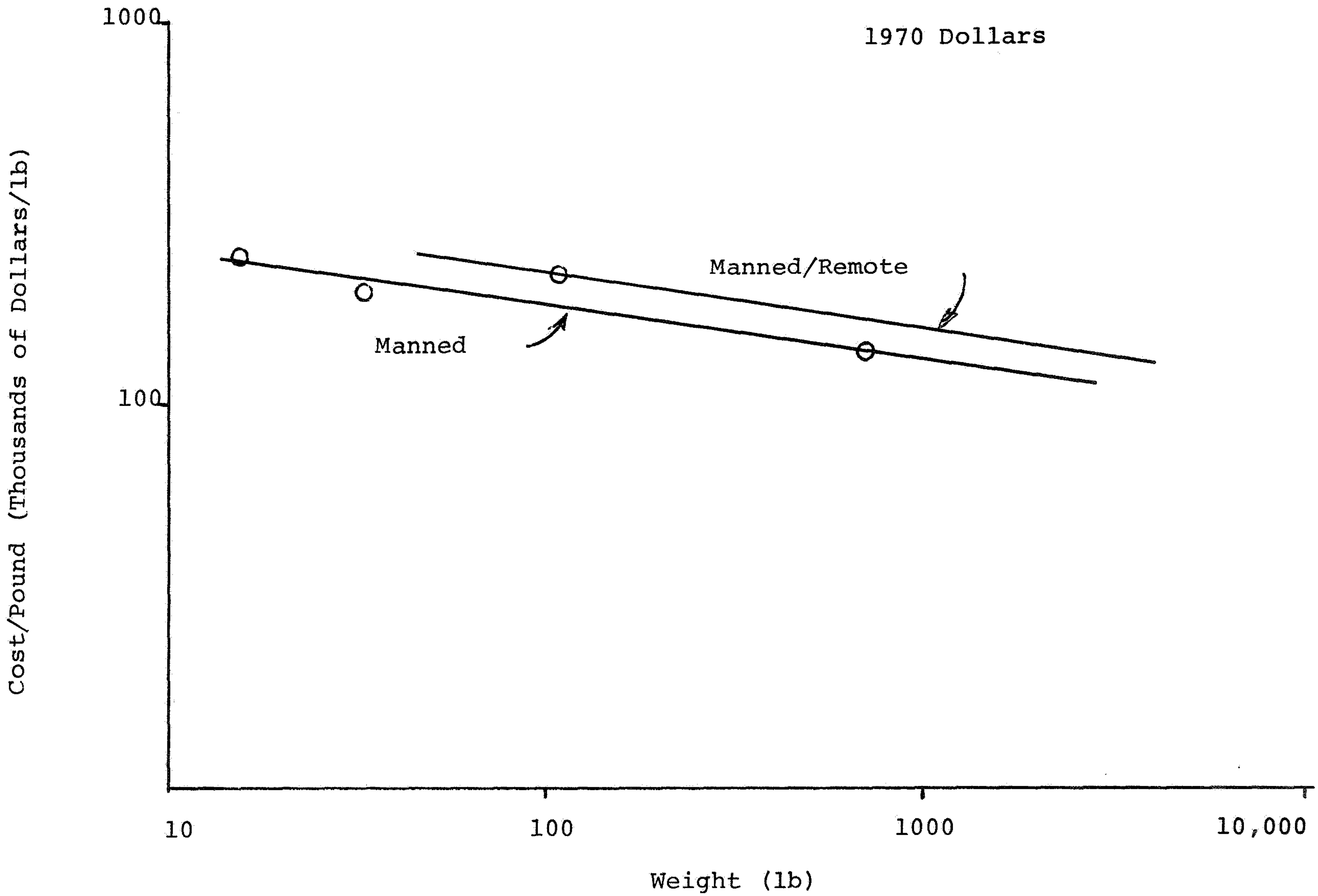

FIGURE 20.2-9 ASTRIONICS RDT&E COST, ROVER CLASS VEHICLES

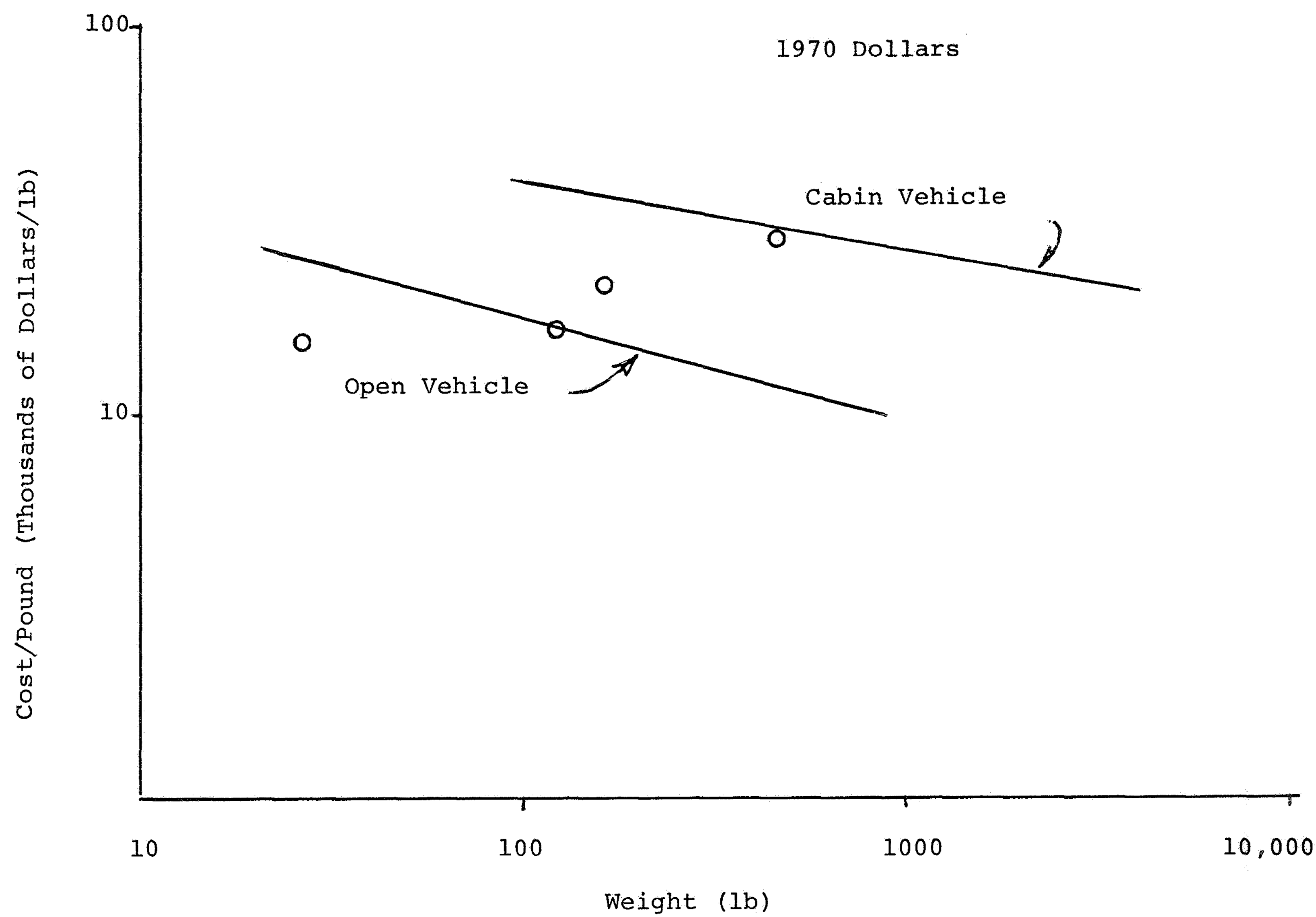

FIGURE 20.2-10 ELECTRICAL POWER RDT&E COST, ROVER CLASS VEHICLE

estimating relationship and the weight from Table 20.3-1. Table 20.3-2 shows the resulting first unit cost and RDT&E cost for each category. It further shows the estimates of the additional cost for integrating the sybsystems, which are also based on historical factors. The total first unit cost and RDT&E are thus estimated to be 26.8 and 458 million dollars, respectively.

The final portion of this chapter presents a summary of the calculations for the synthesized vehicle involved in determining the cost estimate for each of the cost factors defined in Chapter 9. The cost estimate for RDT&E has, of course, been determined. The vehicle production cost estimate, for a learning curve exponent of 1.0, is $(15)\ (26.8 \times 10^6) = 402 \times 10^6$.

TABLE 20.3-1 DISTRIBUTION OF COMPONENT WEIGHTS TO COST CATEGORIES

Mobility System Component	Structure (lb)	Locomotion (lb)	Crew Station (lb)	Astrionics (lb)	Electrical Power (lb)
Astrionics & Manipulators	82			418	
Batteries, Fuel Cells, RTG's					460
Cabin	564		986		
Cryogenic Tanks	800				
Idlers		80			
Motors		260			
Radiators					85
Structure	950				
Suspension		320			
Tracks		500			
Wheels		480			
Hardware Weight (lb) for each category:	2,396	1,640	986	418	545

TABLE 20.3-2. CALCULATED COST ESTIMATES FOR THE MULE

(Thousands of Dollars)

	WEIGHT	FIRST UNIT COST	RDT &E COST
Structure	2396	$4,992	$69,320
Locomotion	1640	3,444	36,180
Crew Station	986	7,220	89,900
Astrionics	418	6,635	71,400
Electrical Power	545	872	13,750
Subtotals	5985	$23,163	$280,550
Tooling, GSE, Integration Trainers, Etc.		3,618	177,511
Totals		$26,781	$458,061

The deployment cost, based on 15 vehicles have a gross weight of 9,005 lb. each, can be approximated by

$$(15)\ (9005\ \text{lb})\ (\$1500/\text{lb}) = \$203 \times 10^6$$

Tables 20.3-3 and 20.3-4 were prepared from data (range/mission and astronaut activity time/mission) generated during the evaluation of the effectiveness of the synthesized mobility system in Section 9.8 of this report. Table 20.3-3 shows that for the ten year period 1980-1990, the projected total distance traversed by the synthesized mobility system in performing the specified missions to its designed capability would be 239,560 km. The corresponding total astronaut activity time would be 18,152 hrs.

TABLE 20.3-3. TOTAL DISTANCE TRAVERSED

Mission	Range/Mission (KM/Mission)	Number of Missions	Range Traversed (KM)
M1	1500	50	75000
M2	1400	50	70000
M3	250	192	48000
M4	250	192	48000
M5	5	190	950
M6	10	18	180
M7	300	7	2100
M8	6	18	108
M9	6	19	114
M10	6	18	108
			239,560 KM

TABLE 20.3-4. TOTAL MANNED MISSION ACTIVITY TIME

Manned Mission	Activity Time/Mission (Hr/Mission)	Number of Missions	Activity Time (Hrs)
M3	52	192	9984
M4	32	192	6144
M5	8	190	1520
M6	24	18	432
M8	2	18	36
M10	2	18	36
			18,152 HRS

The design of the synthesized mobility system was such that vehicle locomotion required an average expendure of 2.5 lb/km. The crew expended consumable at a rate of 2 lb/hr. Thus, the total weight of consumables would be:

Crew:	36,300	lb.
Locomotion :	623,900	lb.
	660,200	lb.

At the specified rate for lunar deployment of \$1500/lb, the operating cost for providing the necessary consumables for the synthesized mobility system to perform the 1980-1990 lunar surface activities would be

$$(660{,}200 \text{ lb}) (\$1500/\text{lb}) = \$991 \times 10^6$$

Finally, with manned mission activity time specified to be \$100,000/hr., then the operating cost for crew participation in performing the Lunar surface activities would be

$$(18{,}152 \text{ hrs.}) (\$100{,}000/\text{hr}) = \$1{,}816 \times 10^6$$

The results of these calculations are summarized in table 9.7-1 of this report, the comparative cost estimate for the synthesized mobility vehicle of this study being \$3,870 million dollars.

REFERENCES

1. "Mimosa Planning Methodology", Vol. 1, Lockheed Missiles and Space Corporation Final Report, April 1967

2. "Preliminary Design Study of a Lunar Local Scientific Survey Module". The Boeing Company, Space Division, Final Technical Report D2-83012-1, June 1966

3. "Feasibility Investigation of a Lunar Mobility Concept", TRW, Systems Group, Volume 1, Technical, May 1969

4. "Study of One Man Lunar Flying Vehicle", Bell Aerosystems Company, Final Report 7335-950010, July 1969

5. "A Graphical Costing Procedure For Advanced Manned Spacecraft" Program Resources Division, Operations Analysis Branch, NASA TM X-58012, October 1967. (Updated to 1970 Dollars)

6. "The Space Planner's Guide", Planning Research Corporation, NASA CR-61180, July 1965

7. "Dual Mode Lunar Roving Vehicle", Grumman Aerospace Corporation, Final Report, Volume II, Book I, February 1970

CHAPTER 21

THERMAL ANALYSIS

Charles H. Byers

The hostile thermal environment of the moon, which has already been discussed in the chapter on lunar characteristics, makes an adequate thermal control system absolutely necessary for the satisfactory performance of a manned or unmanned mobility system. By its nature the study of the thermal properties of a vehicle must follow the development of the basic concept. Since the overall study has been a conceptual one, thermal analysis was not developed in any detail. However, some recommendations have been formulated concerning the general characteristics of such a system without actually performing the detailed numerical design. With time permitting, this latter study would have been pursued as a first priority item. The chapter is subdivided into discussions of the cabin cooling sub-system, the power supply and mechanical cooling sub-system and the heat transfer problems associated with the cryogenic storage tanks.

21.1. Cabin Cooling Loop

Since the cabin will only be in use during manned missions and since it is deemed likely that it will be preferable to perform these missions primarily during lunar day, it is likely that the cabin cooling will be more critical than will be heating. However, in order to cover all contingencies, the potential for lunar-night operation must be fully accounted for, and hence, heating is included. A schematic diagram of the cabin cooling system is given in Figure 21.1-1. The heat transfer medium is an aqueous ethylene glycol solution. Its selection is based upon the fact that a eutectic mixture which contains approximately 40 mole % glycol will remain in the liquid state over a considerable temperature range. (-60°C to 180°C) In addition, it retains acceptable heat transfer and viscous properties throughout this range. A pump forces the circulation of the fluid through the various portions of the loop.

There are two means of heat removal provided in this system. The first is the space radiator, which is mounted on one side of the track cover. Basically it consists of a large rectangular panel shaped somewhat like a clap-board wall, so that the orientation of the main surfaces will be skyward. The panels will have the highly emissive zinc oxide coating used on other space structures. Thin coils will be welded to these surfaces which will carry the hot returning coolant. Whether the area of this radiator is sufficient could only be determined by much more detailed thermal analysis. It might be possible to mount additional radiator area on the roof of the cabin if more area were needed. A small coolant surge tank is provided to assure that leakage does not deplete the system.

The second means of removing heat from the cabin loop is through

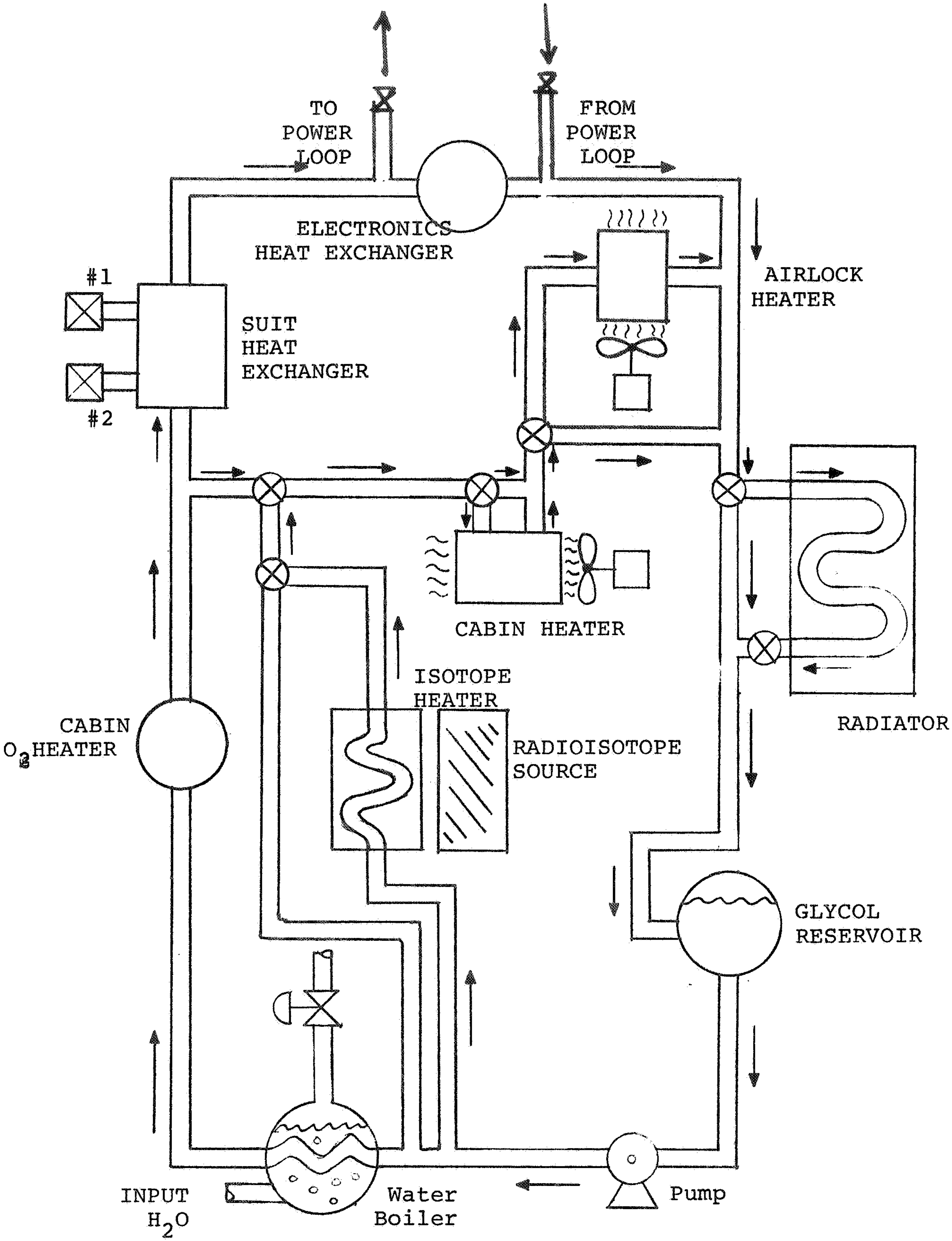

FIGURE 21.1-1. CABIN THERMAL CONTROL LOOP

a water boiler. Water, being a product of the fuel cells, is stored and in periods of intense solar radiation upon the vehicle, water is boiled simply by a controlled leaking of its vapors from a boiler to the lunar environment. The coolant which flows in coils through the water boiler vessel provides the heat of vaporization.

During lunar night, a radioisotope heat exchanger is used to replace the heat lost from the cabin. Basically, this unit consists of a source imbedded in a rectangular tray which is mounted inside the track mount at the front of the right side. Facing it is another frame which is very similar to the radiator design, but which receives the radiant energy from the isotope source and thus provides heating for the coolant.

Finally, since the oxygen entering the cabin comes directly from the cryogenic tanks, some heat must be added to it by the coolant loop.

Several facilities are available for the heating of the cabin and its sub-systems. First the cabin and airlock both have associated with their operation a convective heater which continually circulates heated air through both areas. Each is mounted on the dividing wall between the cabin and airlock and on either side of the door. Sufficient capacity will be required of each heater to control the entire area's temperature in case of emergency. Facilities have been provided for plugging the space suits into the coolant loop in case of a malfunction in the cabin system which would require long-term driving in a fully pressurized suit. These connections are the quick-disconnect type which would be mated to the suit which will be in vogue at that time. Finally, some of the electronic component could be sources of substantial quantities of heat. Coolant would flow through the critical portions of the assembly to assure cool operation.

Unmanned operations present problems in the cabin thermal control. After some deliberation, it is felt by this observer that a cabin should not have an atmosphere during such operations. Rather the parts which must be heated during such missions, that is mainly the electronics, should be placed in one compact area of the vehicle and be heated or cooled by means of imbedded coils.

Thus, conduction will be the main mode of heat transport in this small isolated system. Obviously the mechanical system and the power system would also be required in this mode so that it is logical for the electronics to receive its control from that loop during unmanned operations. Since the loops must have a point of interconnection in order to assure continued operation in the event of a failure in one loop, such as the malfunction of a regulator or a pump, the electronics package appears to be the logical point for the interchange. It is indicated as such in both of the loop diagrams.

21.2. Power and Mechanical Cooling Loop

An independent coolant loop has been proposed for the power system. This is partially due to the very high heat loads which must be absorbed by the coolant loop in association with the operation of the fuel cells (Figure 21.2-1) and partially due to the desirability of having redundant loops. Finally, it should be noted that the power loop operates at a substantially higher temperature level than the other loop.

A pump, a glycol coolant, a radiator and a water boiler are features of this loop which were used in the cabin loop. The schematic diagram of the power loop, Figure 21.1-2, also indicates some differences. The mechanical system, which consists of the hydraulic suspension and the motors which provide the main drive for the vehicle, must all be thermally controlled. This will require extensive tubing and careful design for conductive heat transport. Such details are beyond the scope of this work. It should be noted that the control specifications on these systems are not rigid (-40 to 140 F), and, therefore, design here should be a relatively simple matter.

The other novel feature of the power loop is the fuel cell system. The oxygen and hydrogen gases which are fed to the cells must be heated by the coolant medium. The coolant is then pumped through a water condenser for the product water which leaves the fuel cells as steam. The reaction in the fuel cell is highly exothermic, and hence, the major load in the loop occurs within the fuel cells themselves. Cooling channels carry the coolant through the cells providing the necessary cooling. Finally the loop contains the previously mentioned bypass which serves as electronics thermal control. Crossover is also shown in this loop.

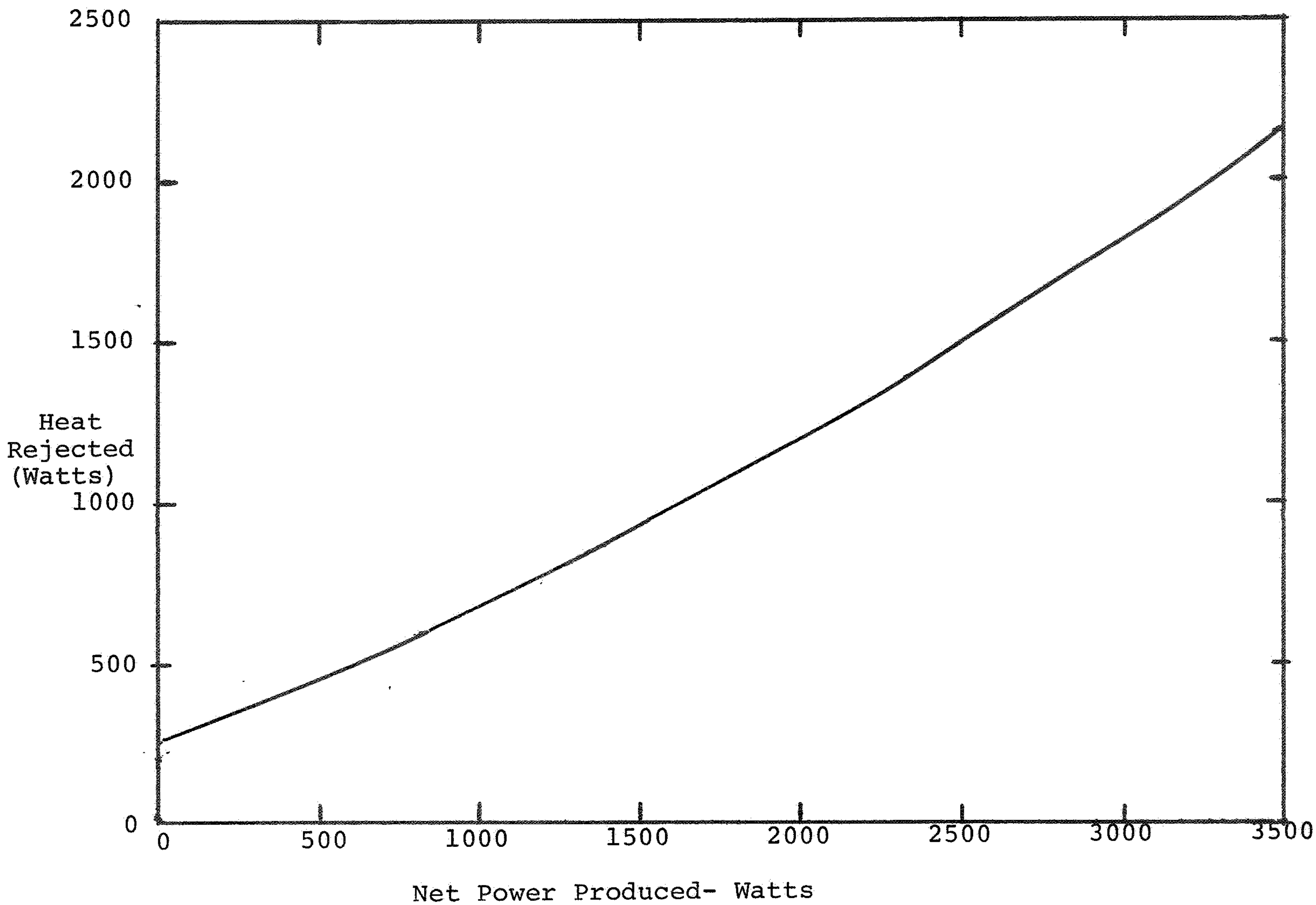

FIGURE 21.2-1 HEAT REJECTION FROM A 28 VOLT ALLIS-CHALMERS FUEL CELL [1]

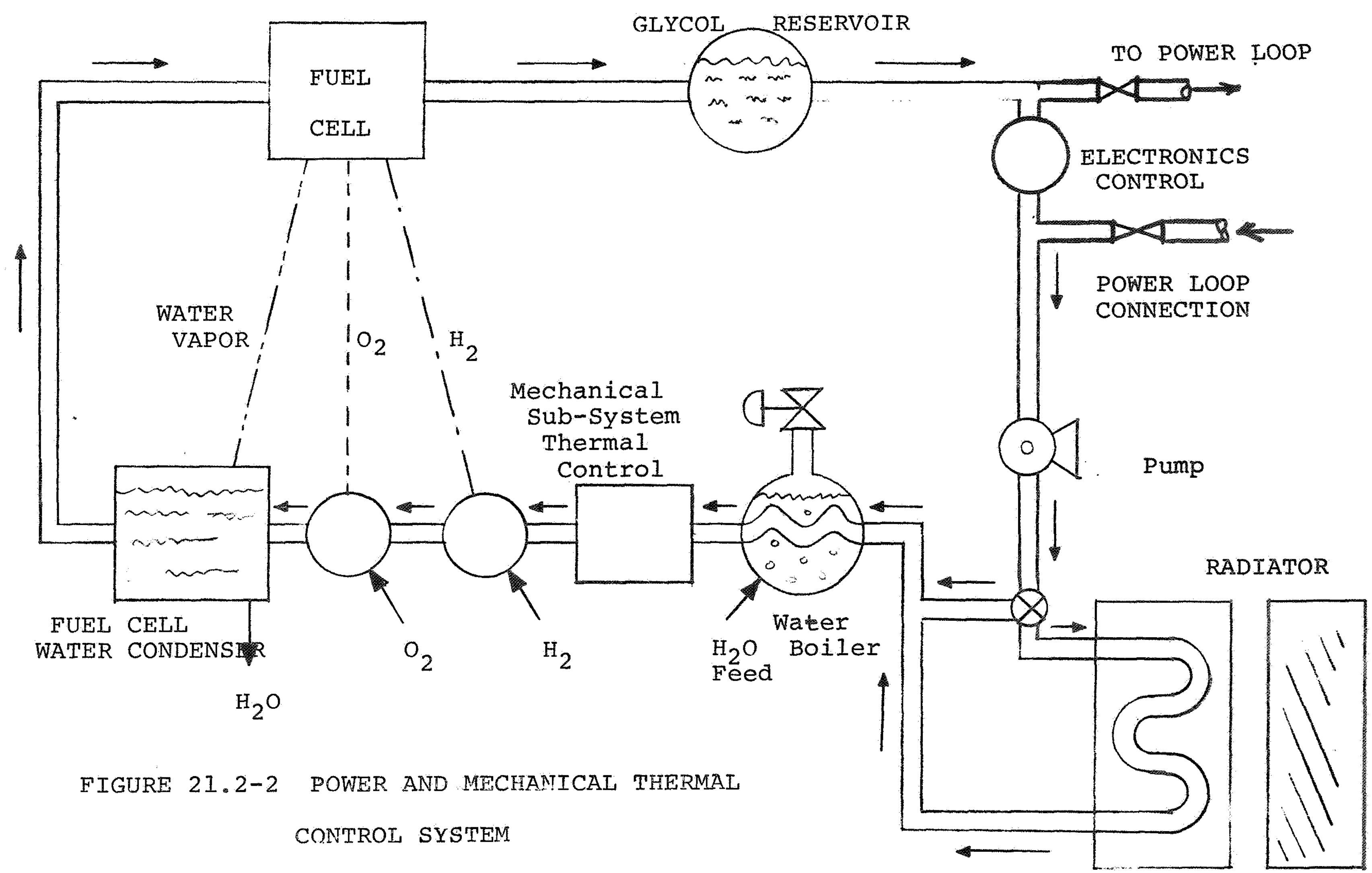

FIGURE 21.2-2 POWER AND MECHANICAL THERMAL CONTROL SYSTEM

21.3. Cryogenics

In the final conceptual design, space was allotted below the payload bay for four cryogenics tanks. Each may be three feet in diameter and five feet in length. No design work was attempted in this area, so that there is no way of being assured that this space is adequate.

In conclusion, it should be repeated that the means by which the problem was approached left little or no time for the consideration of the thermal control sub-system. This task would have some bearing upon the final design and should be one of the first extensions of this study.

REFERENCES

1. Apollo Logistics Support System (ALSS) Payloads. MOLAB, "Environmental Control Subsystems", #D2-83202-4, June 1965.

APPENDICES

APPENDIX A

PRELIMINARY

STATEMENT OF WORK

FOR

THE STUDY OF A MANNED LUNAR-PLANETARY

MOBILITY SYSTEM FOR THE 1980's

This Statement of Work is intended solely for the MSC/University of Houston Summer Design Project and is not an approved area for contractor effort.

SCOPE

This statement of work covers a nine week study, including a one week reporting period, to define a manned, lunar mobility concept to be used in the 1980's. The system will be used to establish a lunar base, conduct long range lunar exploration, and to develop mobility capability for planetary operations. The systems discussed herein do not represent approved programs and this study will not necessarily lead to hardware projects.

INTRODUCTION

One of the major tasks of the National Aeronautics and Space Administration is planning for future space exploration activities. Because of the long leadtimes associated with the development of spacecraft, launch vehicles, and related systems, it is necessary to study potential future missions and their requirements.

The NASA has sponsored various studies of mobility systems for manned and unmanned lunar operations. These studies have covered a wide variety of developments, have contributed to an understanding of operational requirements and have resulted in various vehicle conceptual designs. Future space activities by the NASA may require the development of a new mobility system to support manned lunar operations in the early 1980's. This study is to examine several configurational concepts and operational approaches to a versatile mobility system.

OBJECTIVES

The objectives of this study are:

1. Derive a system concept, which includes development and operational requirements for a mobility system for manned operations in the early 1980's, (which will maximize the commonality with a manned planetary exploration in the late 1980's).* The major design approaches will consider:
    a. All classes of mobility devices such as, rovers, flyers, ground effects machines, and hopping vehicles.
    b. Unmanned operations to support both lunar base development and exploration.
2. Establish requirements for research and technology development, and discuss the potential benefits in terms of system sensitivities, development risks, etc.

GUIDELINES AND ASSUMPTIONS

The following basic assumptions shall be used in the performance of this study:

1. The vehicle will be operational in the lunar environment in the post 1980 period.
2. The range of the vehicle will be (500-750 km manned)* and 1000-1500 km unmanned.
3. The payload range of interest will be 400 lb per man for EVA and 200 lb man for shirtsleeve operations plus 1000-2000 lb of science with delta capability to be determined for unmanned operation.
4. The vehicle life-time should be a minimum of one year, with

a (maximum manned mission of 14 days).*

5. (The vehicle should be capable of operating in the lunar night).*

6. The crew size will be two.

7. The vehicle dry weight will not exceed 5000 lb.

8. The vehicle shall be capable of supporting lunar base construction and extended lunar exploration.

9. Prime mode of delivery of vehicle to lunar surface shall be the lander version of the Space Tug.

10. A lunar space station shall be in existance in lunar orbit at the time of manned surface operations with surface logistics supplied by a lander version of the Space Tug.

11. Single point failures - The mobility system shall provide the completion of the mission with one failure in a sub-system, (and that a second failure shall not endanger the crew).*

12. The mobility system should provide the following capabil-ities:

    a. Climbing and descending slopes of 30 degrees.

    b. Ground clearance of 50 cm.

    c. Obstacle negotiation of 50 cm for a step and 90 cm for a crevasse (for two wheels at zero velocity).*

    d. Stability at pitch and roll angles of 45 degrees.

13. The lunar soil is specified per "MSC Specification Lunar Soil Model" (preliminary), March 10, 1970.

14. The lunar surface roughness is defined in "Lunar Surface Models", by R. E. Hutton, March 1969, TRW Document No. 11468 - 6001 - Ro - 00.

APPROACH

1. The study shall make maximum use of existing documentation on vehicle concepts, science objectives, and operational concepts.
2. Emphasis should be placed on a formalized systems analysis in terms of the following steps.
   a. Functional analysis of mobility system from gross system requirements.
   b. Evaluation of functional analysis to define functional requirements.
   c. Perform trade-off studies to define design requirements.
3. From the standpoint of safety the contractor should place emphasis on simplicity and redundancy in the vehicle.
4. Consideration should be given to the potential commonality of equipment, such as by using remote manipulators in lieu of EVA and also using them for unmanned operations. Also, certain subsystems might be used for remote control and for landing systems if an autonomous landing capability is desirable.

CONTRACTOR'S TASKS

The following outline delineates the essential elements and tasks of this study. It should be understood that expansion and further direction relative to each of these characteristics may be furnished to the contractor as the study progresses.

The study will be conducted in two phases and provide for a midterm review and a final review.

Phase I - Definition of system requirements and selection of the class of vehicle concept to be developed.

Task 1. (5%) The contractor shall supplement existing gross (vehicle)* requirements by the review of current documentation describing operations in the early 1980's within the framework of the Integrated Plan, and other applicable documentation. NASA will review and approve the gross requirements which shall be applicable to the remainder of the study.

Task 2. (10%) Perform and document functional analysis of each candidate mobility system and derive functional requirements from the analysis. (The analysis should be performed to the fourth level).*

Task 3. (10%) Select candidate mobility system concepts and develop to a level consistent with the functional analysis. These mobility systems shall be defined in terms of figures of merit consistent with the functional requirements, and evaluation criteria developed in Task 4.

Task 4. (5%) Establish criteria for the evaluation of the candidate concepts which is based on the functional analysis and requirements.

Task 5. (5%) Utilizing the information previously generated in Task 3 and the criteria developed in Task 4, the contractor shall assemble such charts, tables, and comparison matrices as necessary to support his recommendation of a concept for further development.

Phase II

The concept selected in Phase I is developed in more detail through configuration and subsystem trade-offs and analysis.

Task 6. (45%) Mobility system concept development.

a. The contractor shall prepare such sketches and drawings as necessary to illustrate the mobility system physical configuration and perform design trade-offs (such as integration of a landing system or inclusion of remote manipulators).*

b. The subsystems of the mobility system shall be defined to the extent that their level of performance, state of art, and interfaces with other subsystems can be delineated.

c. Define crew participation, interfaces, and training requirements.

d. Perform a mission operations and science experiments analysis.

Task 7. (10%) Expand the functional analysis and requirements to the level consistent with the configuration and detail of the subsystems. Track and coordinate interfaces to insure subsystem compatibility and perform a failure and effects analysis consistent with the level of subsystem development.

Task 8. (10%) There shall be two formal oral presentations, the midterm briefing and the final program review; in addition, there will be weekly status review meetings between NASA and the study managers. The study shall be documented in a final report which shall contain the data, analysis, drawings, and discussion involved in describing the final mobility system.

* Indicates a subsequent change. See "Revisions in Preliminary Statement of Work".

REVISIONS IN "PRELIMINARY STATEMENT OF WORK"

The following revisions and interpretations were made by MSC during informal discussions.

Add "and Mars night" to statement 5 of Guidelines.

Delete "and that a second failure shall not endanger the crew" from statement 11 of Guidelines.

Delete "for two wheels at zero velocity" from statement 12 of Guidelines.

Replace "vehicle" by "system" in the statement of Task 1.

Delete "such as integration of a landing system or inclusion of remote manipulators" from the statement of Task 6a.

Phase I should occupy 40% of the study and Phase II 60%.

The following groundrule was established: The manned mission is primary and the unmanned mission is secondary.

The lunar mission is of immediate importance with the Mars mission representing growth potential.

Planetary mission may be interpreted as Mars mission.

"Fourth level functional analysis" may be interpreted to mean "such a level to define the subsystems and their interactions".

The mobility system can consist of more than one vehicle.

The launch environment will be specified by MSC.

Maximum g forces during launch will be 3.

Replace "14 days" by "36 hours with a 12 hour contingency" in statement 4 of Guidelines.

Replace "500-750" by "250" in statement 2 of Guidelines.

APPENDIX B
MEMBERSHIP OF AD HOC COMMITTEES

R. Doyle Holstead

The system design team was divided into groups such that each group was identified by the subsystems for which that group was responsible. However, at various times there arose a need for information and/or action that was not related to any single group, and an ad hoc committee was appointed by the project leader to provide the information and/or action. The membership of each committee is listed below.

Team Structure and Organization
Emanuel
Ludeman
Degelman
Shieh
Ulrich

Schedule of Work
Kay
Trieschmann
Chang
Nachlinger
Neathery

Functional Analysis
Shieh
Lawrence
Pikul
Neathery

Configuration Concepts
Leptourgos
Neathery
Pikul
Trieschmann
Yuster

Measure of Effectiveness
Lawrence
Pikul
Lindholm
Yuster
Leptourgos

Initial Functional Analysis
Clark
Byers
Pucacco
Sneckenberger

Evaluation
Sneckenberger
Degelman
Nachlinger
Leptourgos
Clark
Chang

Cost - Effectiveness
Nachlinger
Sneckenberger

Synthesis of Candidate Systems
Byers
Degelman
Leptourgos
Yuster
Pucacco

Subsystem Description
Neathery
Kay
Chang
Ludeman
Shieh
Ulrich

Remote Control Problems
Pikul
Clark
Emanuel
Lawrence

Functional Analysis of Candidates
Degelman
Shieh
Leptourgos
Chang
Neathery

Adequacy Indices for Locomotion
Pucacco
Kay
Clark
Ulrich

Final Configuration
Pucacco
Emanuel
Clark
Leptourgos
Sneckenberger
Degelman

Final Presentation
Yuster
Kay
Neathery
Lawrence
Pikul
Byers

Inflated Track Candidate
Degelman
Neathery
Chang

Track Candidate
Kay
Leptourgos
Lawrence

Half-Track Candidate
Sneckenberger
Nachlinger
Shieh

Appendix C

ENCLOSED LUNAR TRANSPORTATION SYSTEM (ELTS)

George Trieschman

A concept is proposed which would produce a life supporting environment on the lunar surface from material found primarily on the moon. A tubular structure is envisioned to allow access from Moonlab to areas of continuing scientific interest and space-tug landing sites.

The concept will require that heat processing of lunar surface fines be developed so that these fines can be melted and combined to produce a continuous tube capable of safely sustaining human life. Since lunar fines contain 20% glass[1] it would seem that a spun glass sturcture generated by a slowly moving surface vehicle is most promising. The process would develop from technology already in use to produce wound glass casings for Polaris and Minuteman missiles[2]. Modification of this process for extra-terrestrial use will require considerable research and development during the 1970-1980 time period.

---

[1] Perna, A. J., "Feasibility of Melting Lunar Soils", NASA/ASEE Memo, Houston, Texas, August 1970.

[2] Gatzek, L. E., "Advanced Space Structures Technology", NASA/ASEE Summer Faculty Institute Seminar, Houston, Texas, 1967, pp. 418-428.

APPENDIX D

FUNCTIONAL ANALYSES OF

TYPICAL MISSIONS

Raymond F. Neathery

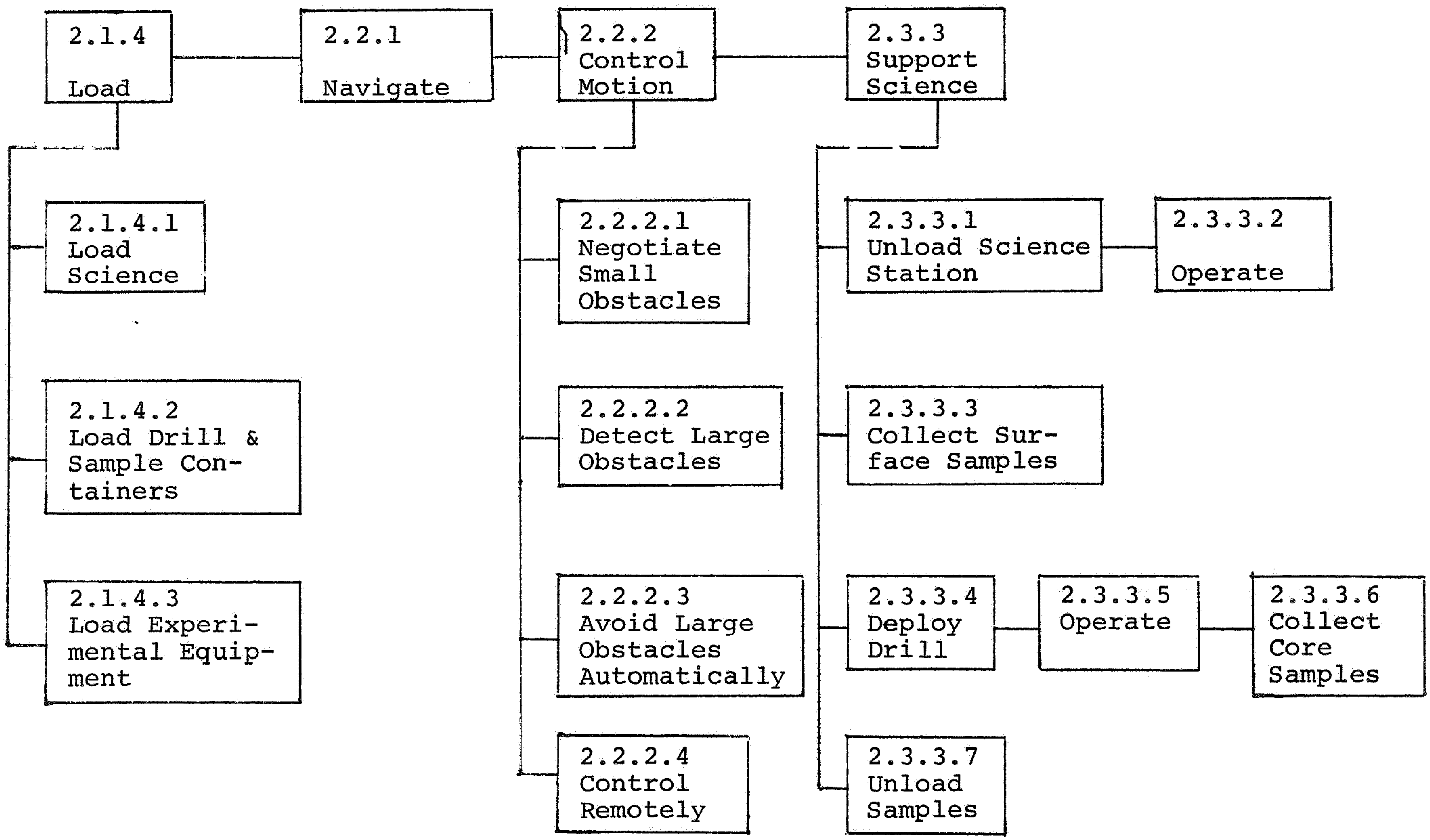

FIGURE D-1 FUNCTIONAL ANALYSIS OF UNMANNED GEOLOGY-GEOPHYSICS MISSION

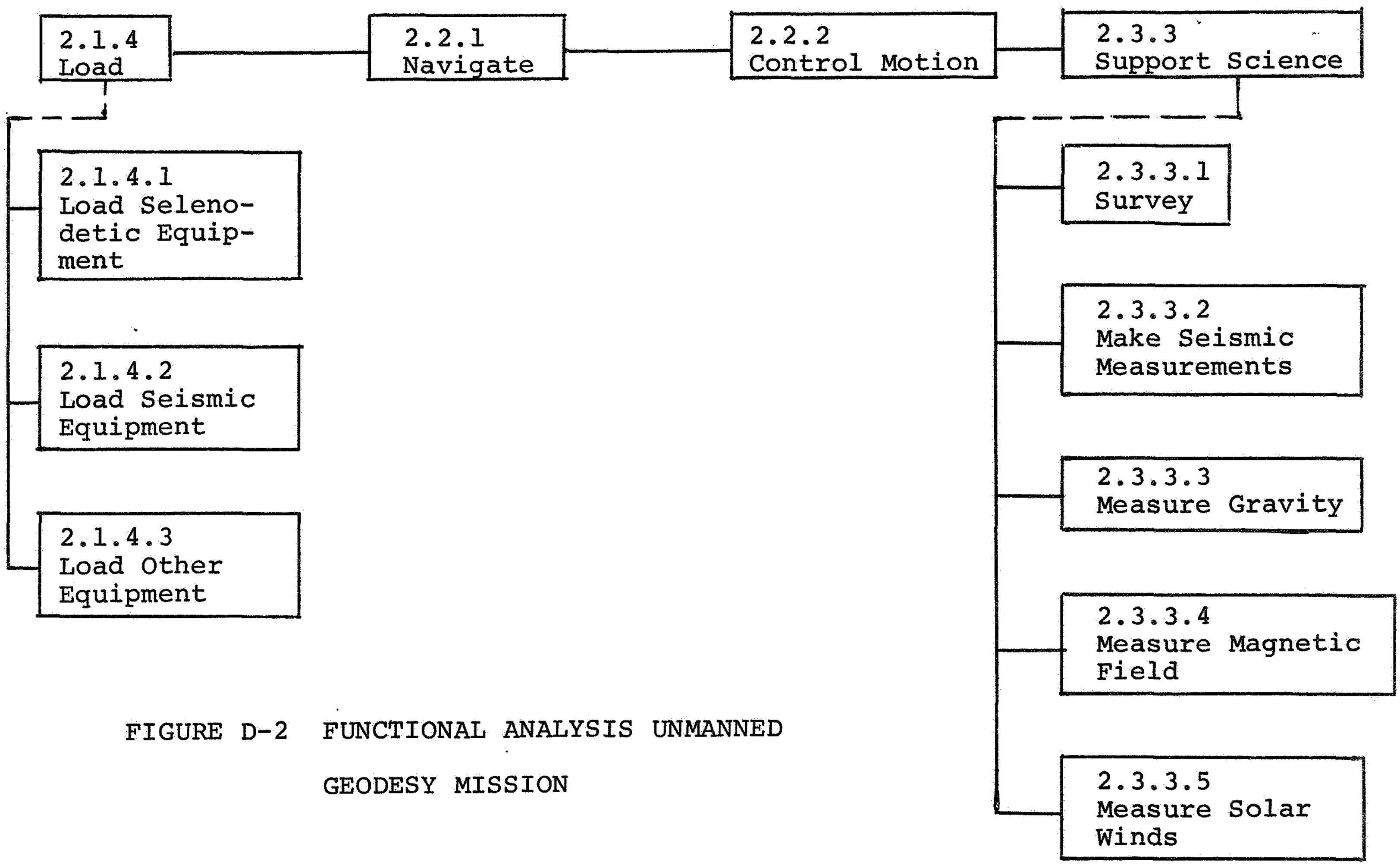

FIGURE D-2 FUNCTIONAL ANALYSIS UNMANNED

GEODESY MISSION

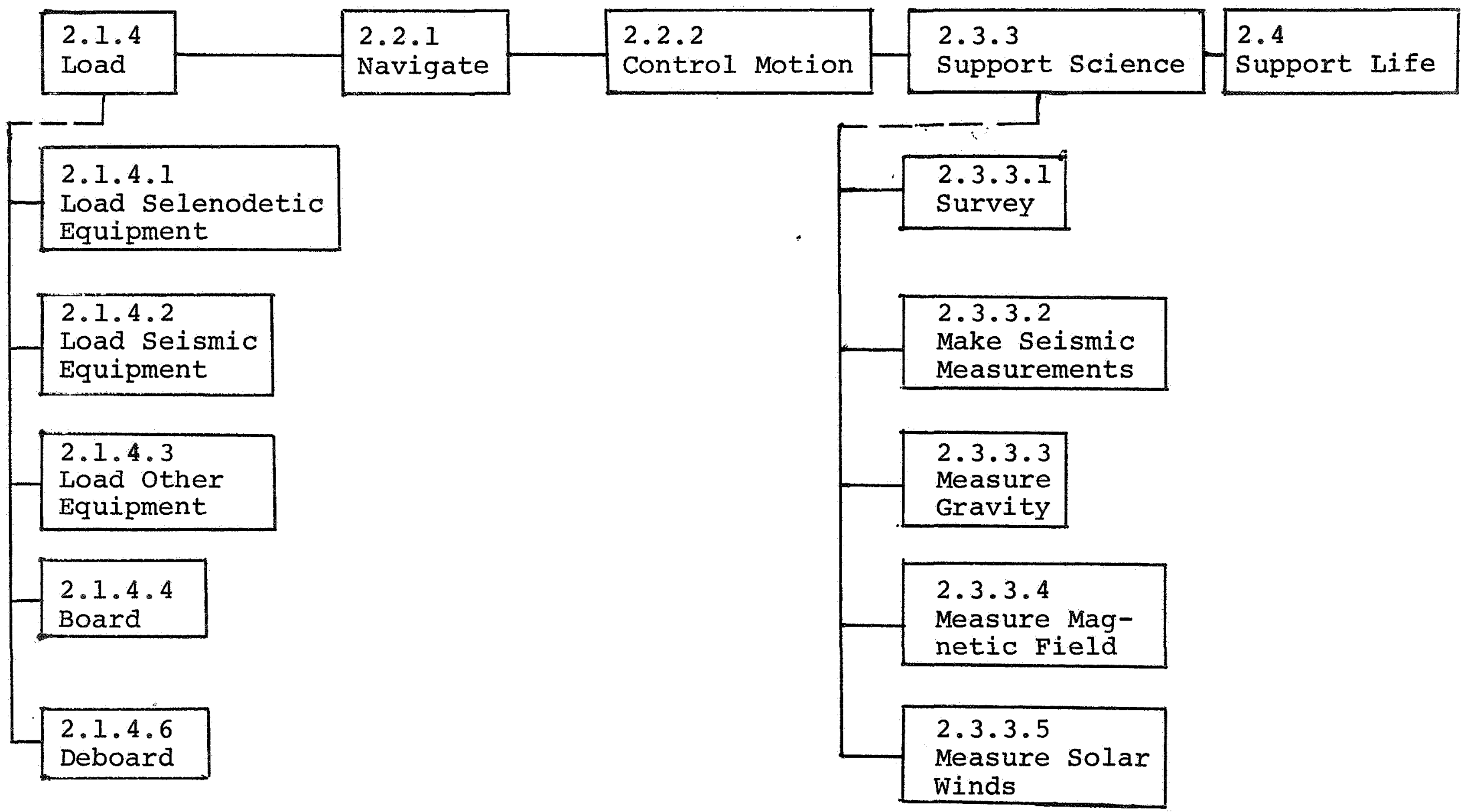

FIGURE D-3 FUNCTIONAL ANALYSIS MANNED GEODESY MISSION

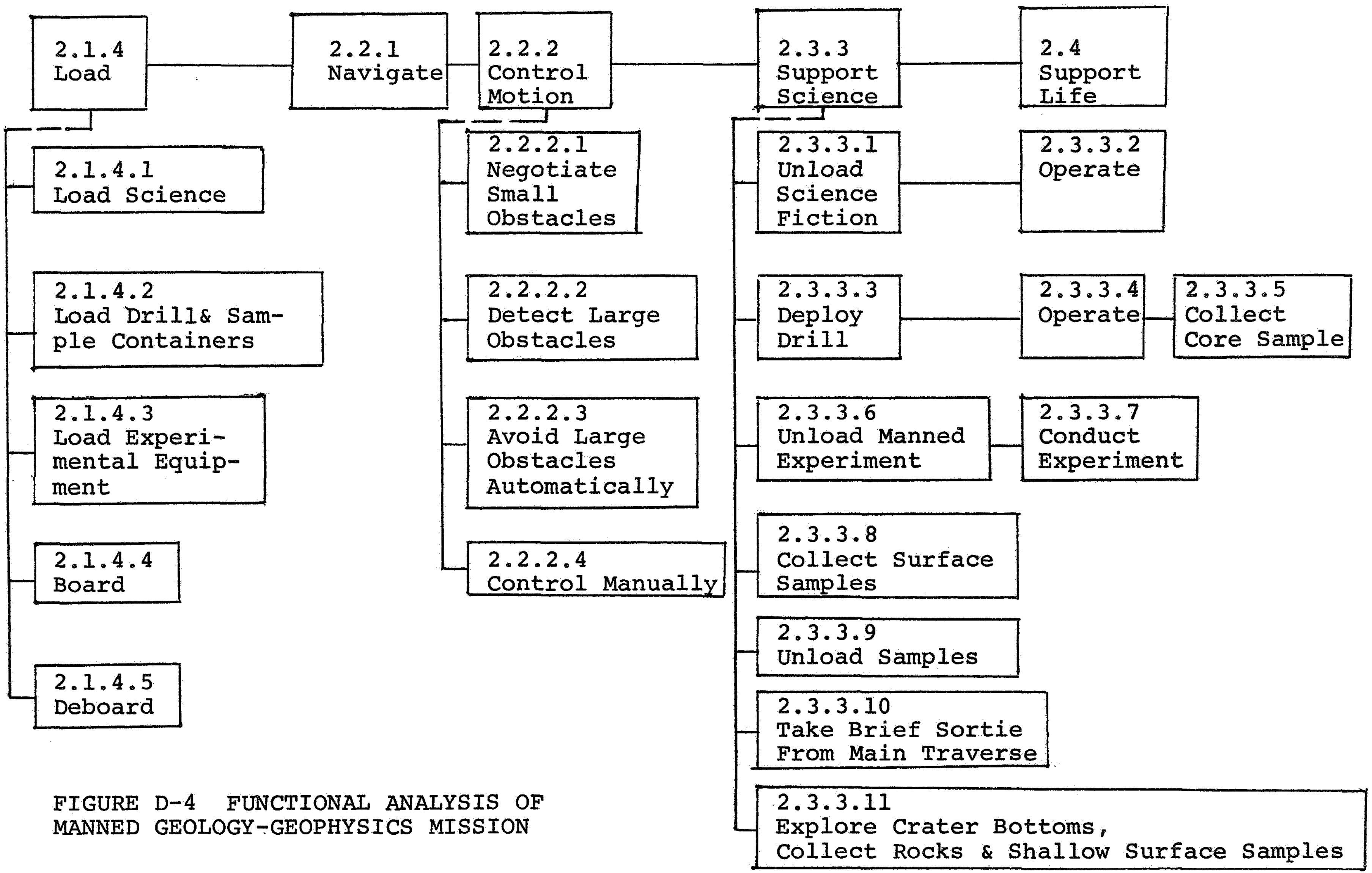

FIGURE D-4 FUNCTIONAL ANALYSIS OF
MANNED GEOLOGY-GEOPHYSICS MISSION

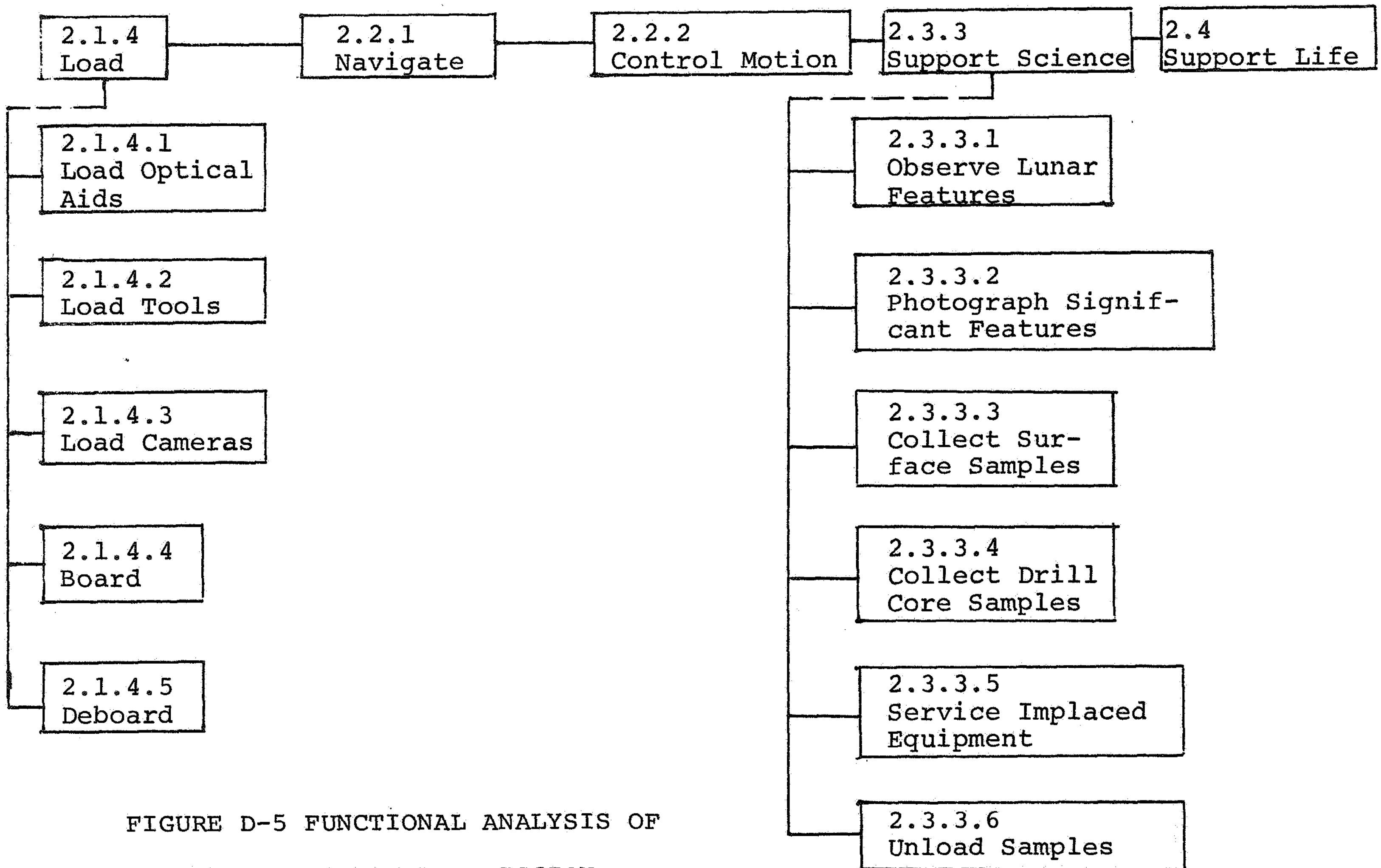

FIGURE D-5 FUNCTIONAL ANALYSIS OF MANNED INSPECTION MISSION

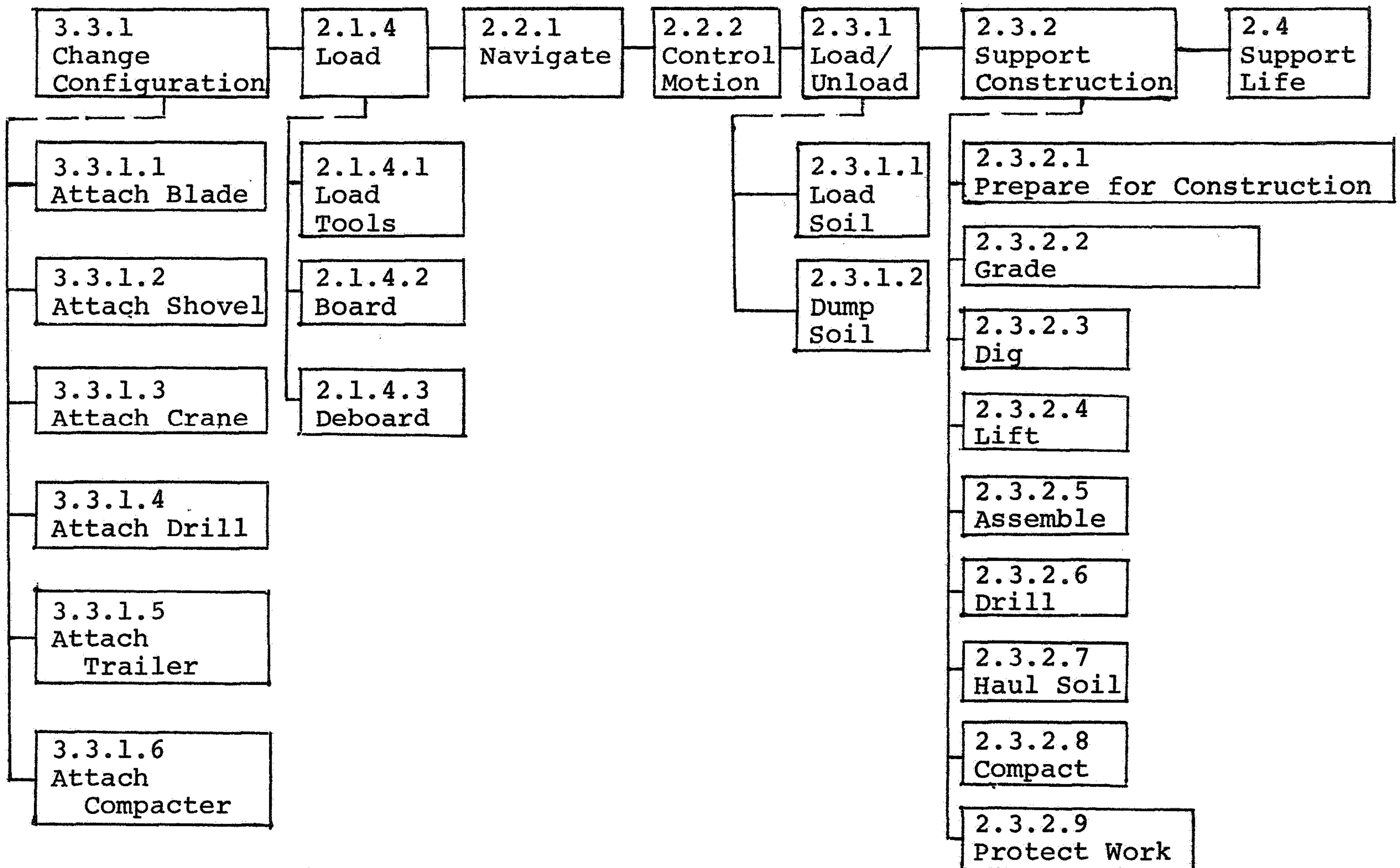

FIGURE D-6 FUNCTIONAL ANALYSIS OF

MANNED CONSTRUCTION MISSION

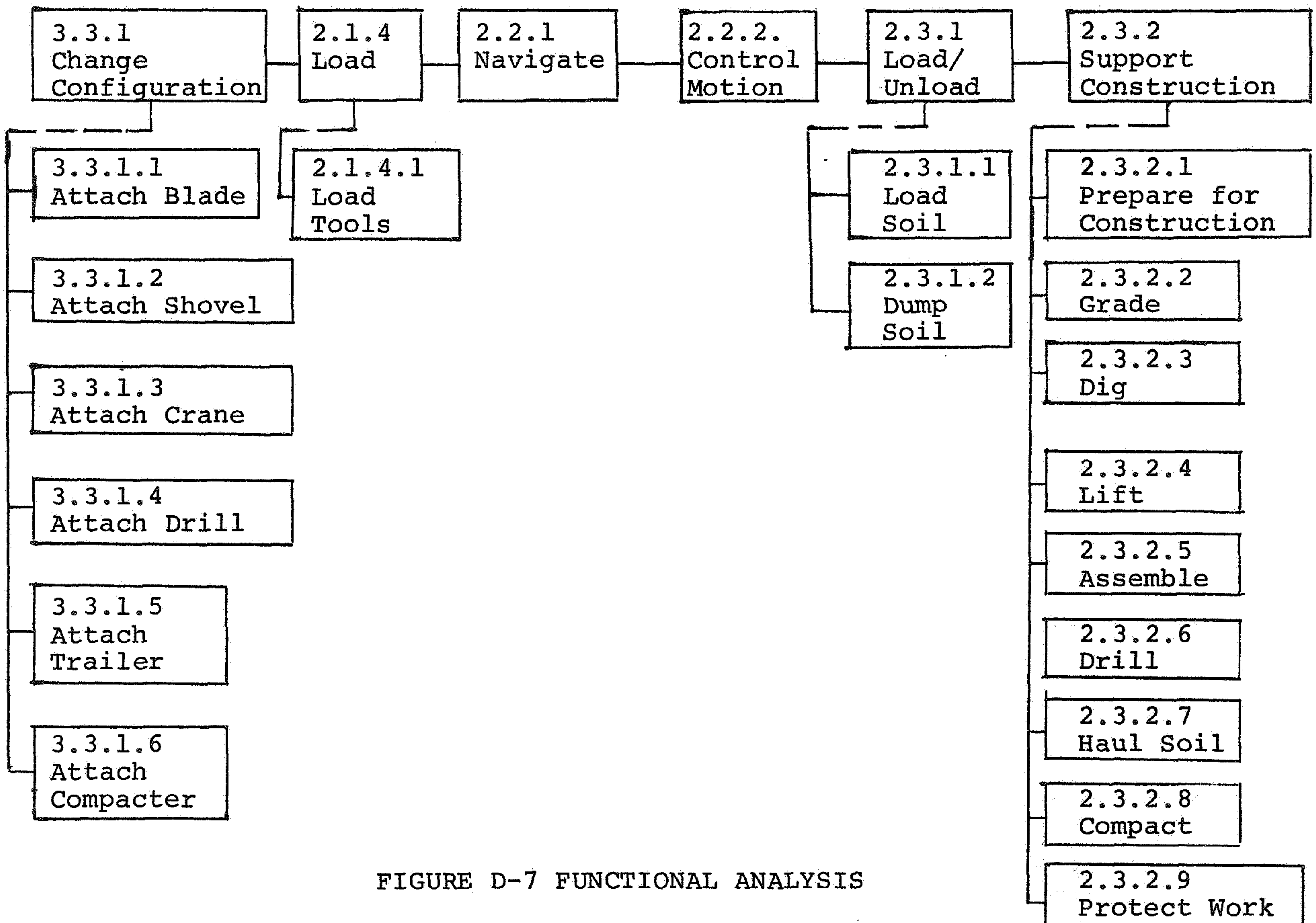

FIGURE D-7 FUNCTIONAL ANALYSIS

OF UNMANNED CONSTRUCTION MISSION

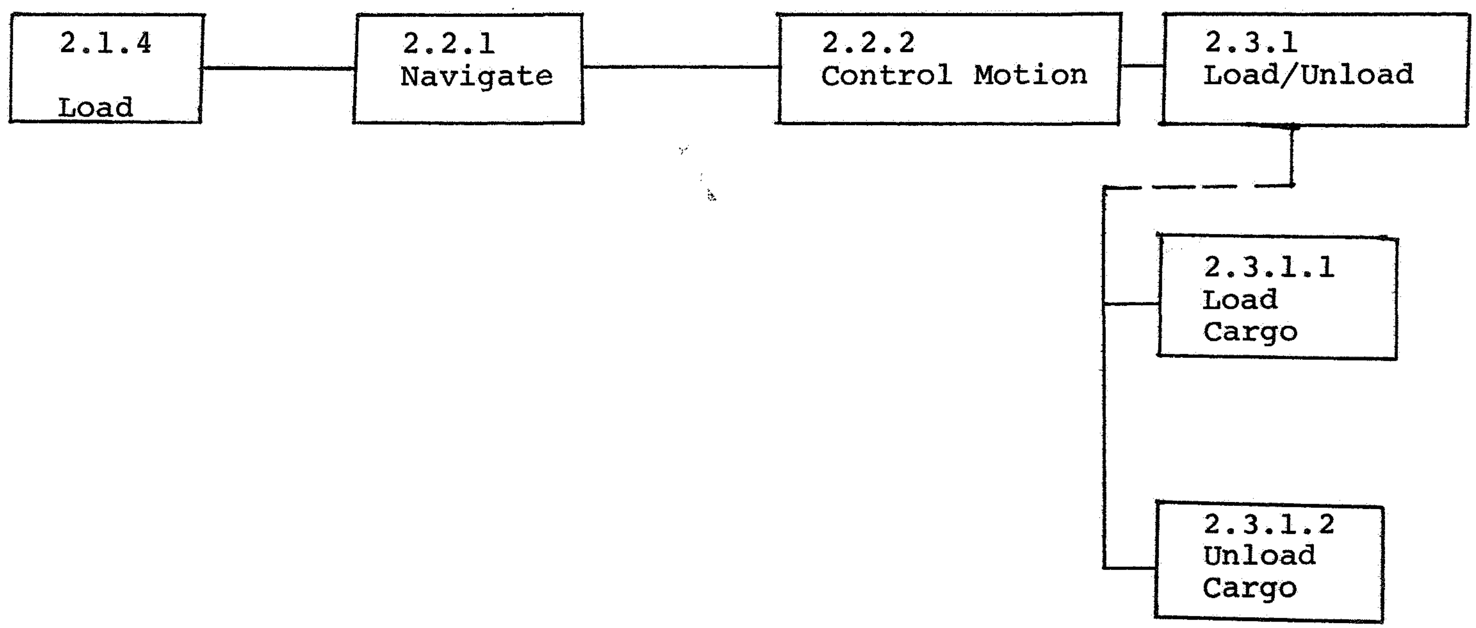

FIGURE D-8 FUNCTIONAL ANALYSIS OF
UNMANNED SUPPLY MISSION

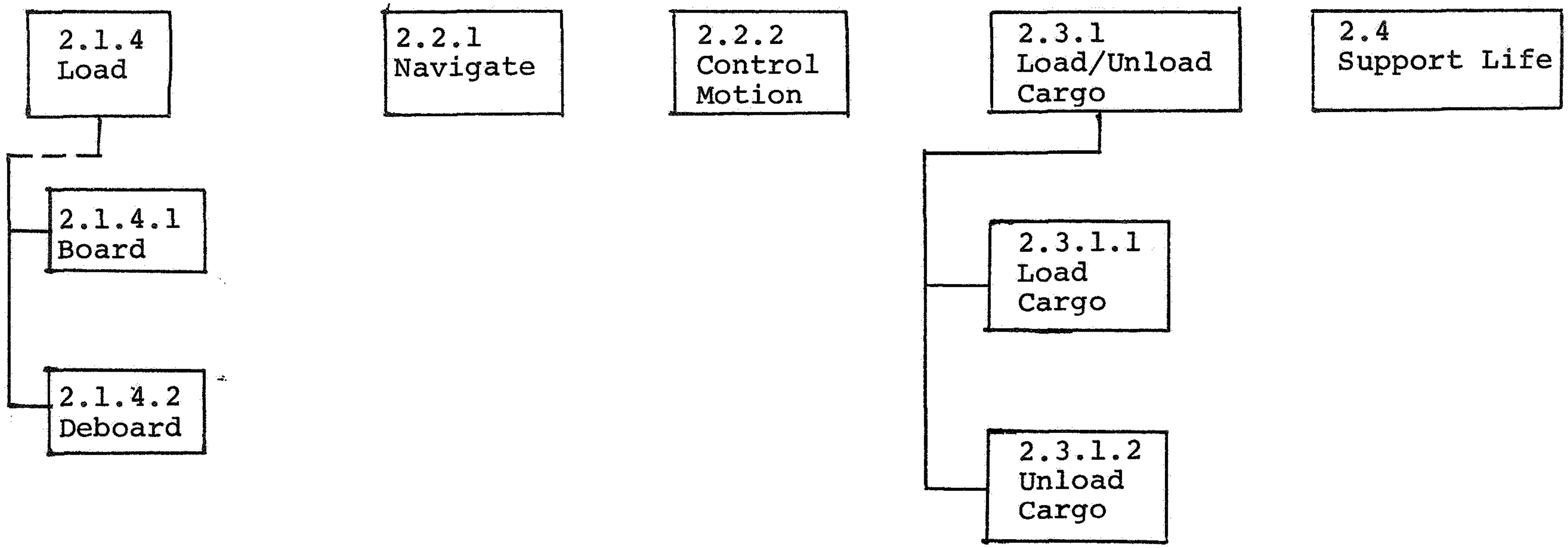

FIGURE D-9 FUNCTIONAL ANALYSIS OF

MANNED SUPPLY MISSION

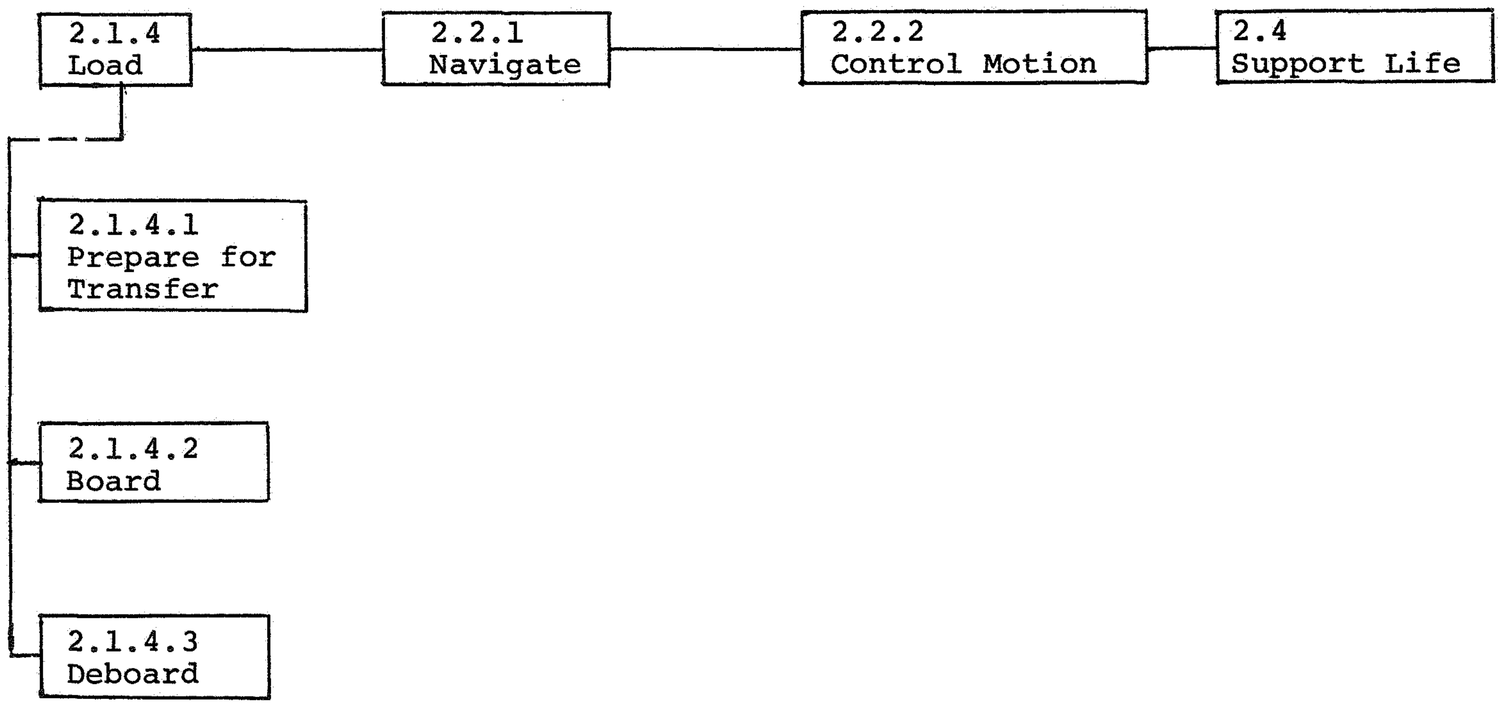

FIGURE D-10 FUNCTIONAL ANALYSIS OF HUMAN TRANSPORT MISSION

APPENDIX E

EFFECTIVENESS - COST COMPUTER PROGRAM

Charles H. Byers

In anticipation of the task of calculating the effectiveness and cost of a large number of candidate mobility systems a computer program was written to execute these computations. This section gives basic information on the computer program, a copy of the source printout, and information on the method of utilization of the program.

D.1. Program Information

The program was written to execute the computations described in Chapter 9. FORTRAN V computer language was utilized, with the program being suitable for execution on the UNIVAC 1108 computer at the Manned Spacecraft Center. Special attention was paid the output formats to assure their ease of readability and access of information. Figure D.1-1 is the source program generated for the study. All terms are defined by comment cards.

D.2. Required Input

Twelve cards are required for the execution of the cost-effectiveness program. They are the following: (cc refers to card column).

```
C     PROGRAM FO THE DETERMINATION OF EFFECTIVENESS AND COST OF CANDID-
C     ATE MOBILITY SYSTEMS - - NASA-ASEE SYSTEMS DESIGN INSTITUTE 1970

C     GLOSSARY OF IMPORTANT SYMBOLS
C     ** POT(N,J) ** POTENTIAL OF SYSTEM J FOR PERFORMING MISSION TYPE N
C     ** WT(N) ** WEIGHTING FACTOR ASSIGNED TO MISSION TYPE N
C     ** ANO(N) *** NUMBER OF TYPE N MISSIONS
C     ** EFF(J)  ** EFFECTIVENESS (ABSOLUTE) OF SYSTEM J
C     ** PCEFF(J) ** PERCENT EFFECTIVENESS OF SYSTEM J
C     ** TCOST(J) ** TOTAL COST FOR SYSTEM J
C     ** WEIGHT(K) ** DRY WEIGHT (LB) OF SYSTEM(K)
C     ** DEPLOY(K) ** NUMBER OF K-TYPE SYSTEMS WHICH ARE DEPLOYED
C     ** SUBWT(L,K) ** WEIGHT OF SUBSYSTEM L OF SYSTEM K
C     ** CONSUM(L,K) ** WEIGHT OF CONSUMABLES REQ'D ON SYSTEM K IN ORDER
C                       TO PERFORM MISSION L
C     ** ASTIME(L,K) ** HOURS REQ'D BY MAN TO PERFORM MISSION L USING
C                       SYSTEM K
C     ** CWT(K) ** CONSUMABLES TOTAL WT FO ALL MISSIONS USING SYSTEM K
C     ** SUBCOS(L,K) ** COST OF SUBSYSTEM L OF SYSTEM K
C     ** RDCOST ** SINGLE UNIT R + D COSYS FOR SYSTEM K
C     ** DEVCST(K) ** TOTAL DEVELOPMENT COST FOR SYSTEM K
C     ** COSMAN ** COST PER MAN- HOUR OF LUNAR STAY TIME
C     ** B(K) ** NUMBER OF PROTOTYPES OF SYSTEM K
C     ** COSLB ** COST TO DEPLOY A POUND PF ANY SYSTEM ON THE MOON
C     ** DEVEXP ** DEVELOPMENT EXPONENTIAL
C     ** CSTOP(K) ** OPERATING COST
C     ** TWT(K) TOTAL WEIGHT OF SYSTEM K
C     ** TAT(K) ** TOTAL ASTRONAUT TIME ASSOCIATED WITH SYSTEM K
C     ** FWT(K) ** TOTAL FIXED WEIGHT OF THE SYSTEM K
```

FIGURE D.1-1 EFFECTIVENESS - COST COMPUTER PROGRAM

```
      DIMENSION TWT(5),SUBWT(50,50)   ,POT(50,50),EFF(50),WT(50),ANO(50),
     1PCEFF(50),RDCOST(50),TAT(50),CWT(50),FWT(50),DEVCST(50),CSTOP(50),
     2CSTMAN(50),TCOST(50),ASTIME(50,50),SUBCOS(50,50),CONSUM(50,50),
     3WEIGHT(50),DEPLOY(50)
      DIMENSION B(50)
      JO=0
      WRITE(6,200)
  200 FORMAT(1H , 41X,50H NASA-ASEE SYSTEMS DESIGN INSTITUTE 1970
     1      )
      WRITE(6,201)
  201 FORMAT(1H ,47X,40H EFFECTIVENESS-COST STUDIES
      READ(5,100) NRUN
  100 FORMAT(I3)
    1 JO=JO+1
      READ(5,101) J,N,NO,COSLB ,COSMAN,DEVEXP
  101 FORMAT(3I3,2E10.4,F5.4)
      READ(5,102) ((POT(L,K),L=1,N),K=1,J)
  102 FORMAT(20F4.3)
      READ(5,104) (WT(K),K=1,N)
  104 FORMAT(20F4.3)
      READ(5,105) (ANO(K),K=1,N)
  105 FORMAT(20F4.1)
      READ(5,110) (WEIGHT(K),K=1,J)
  110 FORMAT(16F5.0)
      READ(5,111) (DEPLOY(K),K=1,J)
  111 FORMAT(20F4.0)
      READ(5,125) ((SUBWT(L,K),L=1,NO),K=1,J)
```

FIGURE D.1-1 EFFECTIVENESS - COST COMPUTER PROGRAM (CONTINUED)

```
125 FORMAT(16F5.0)
    READ(5,126) (( CONSUM(L,K),L=1,N ),K=1,J)
126 FORMAT(16F5.1)
    READ(5,126) (( ASTIME(L,K),L=1,N ),K=1,J)
    READ(5,103) (( SUBCOS(L,K),L=1,NO),K=1,J)
103 FORMAT(8E10.4)
    READ (5,106) (B(K),K=1,J)
106 FORMAT(16F5.0)
C   WRITES OUT THE INPUT EFFECTIVENESS FACTORS
    IF(JO.GT.1) GO TO 20
    WRITE(6,202) JO,J,NO,N
202 FORMAT(1H0, 11HRUN NUMBER ,I3,4X,29HNUMBER OF SYSTEMS CONSIDERED ,
   1I3,4X,32HNUMBER OF SUBSYSTEMS PER SYSTEM ,I3,4X,24HNUMBER OF MISSI
   2ON TYPES ,I3)
    GO TO 21
 20 WRITE(6,212) JO,J,NO,N
212 FORMAT(1H1, 11HRUN NUMBER ,I3,4X,29HNUMBER OF SYSTEMS CONSIDERED ,
   1I3,4X,32HNUMBER OF SUBSYSTEMS PER SYSTEM ,I3,4X,24HNUMBER OF MISSI
   2ON TYPES ,I3)
 21 WRITE(6,203)
203 FORMAT(1H0,38X,43HINPUT SYSTEM-MISSION EFFECTIVENESS FACTORS  )
    WRITE(6,204)
204 FORMAT(1H0,94HSYS    MSSN 1    MSSN 2    MSSN 3    MSSN 4    MSSN 5    M
   1SSN 6    MSSN 7    MSSN 8    MSSN 9    MSSN 10   )
    DO 23 K=1,J
    WRITE(6,205) K,(POT(L,K)  ,L=1,10)
205 FORMAT(1H0,I3,10(4X,F5.3))
 23 CONTINUE
    WRITE(6,206) (WT(L),L=1,10)
206 FORMAT(1H0,3HWTS,10(4X,F5.4))
    WRITE(6,207) (ANO(L),L=1,10)
207 FORMAT(1H0,3HANO,10(4X,F5.1))
C   TOTALS OF WEIGHTING FACTORS
```

FIGURE D.1-1 EFFECTIVENESS - COST COMPUTER PROGRAM (CONTINUED)

```
      WTM=0
      DO 4 K=1,N
      WTM=WTM+WT(K)
    4 CONTINUE
      WTM=   1.0*WTM
C     COMPUTATIONS OF EFFECTIVENESS AND COSTS
      DO 5 K=1,J
      EFF(K)=0
      CWT(K)=0
      RDCOST(K)=0
      TAT(K)=0
      DO 6 L=1,N
      CWT(K)=CWT(K)+CONSUM(L,K)*ANO(L)
      RDCOST(K)=RDCOST(K)+SUBWT(L,K)*SUBCOS(L,K)
      TAT(K)=TAT(K)+ASTIME(L,K)*ANO(L)
      POT(L,K)=POT(L,K)*WT(L)
      EFF(K)=EFF(K)+POT(L,K)
    6 CONTINUE
      FWT(K)=WEIGHT(K)*DEPLOY(K)
      TWT(K)=CWT(K)+FWT(K)
      DEVCST(K)=RDCOST(K)*(DEPLOY(K)**DEVEXP+B(K))
      CSTMAN(K)=TAT(K)*COSMAN
      CSTOP(K)=TWT(K)*COSLB
      TCOST(K)=CSTOP(K)+DEVCST(K)+CSTMAN(K)
      PCEFF(K)=EFF(K)*100./WTM
    5 CONTINUE
C     WRITE OUT EFFECTIVENESS RESULTS
      WRITE(6,208)
  208 FORMAT(1H1,49X,22HEFFECTIVENESS RESULTS )
```

FIGURE D.1-1 EFFECTIVENESS - COST COMPUTER PROGRAM (CONTINUED)

```
      WRITE(6,209)
  209 FORMAT(1H ,27X,30HWEIGHTED EFFECTIVENESS FACTORS,38X,1H/,3X,19HTOT
     1AL EFFECTIVENESS    )
      WRITE(6,210)
  210 FORMAT(1H ,94HSYS    MSSN 1    MSSN 2    MSSN 3    MSSN 4    MSSN 5    M
     1SSN 6    MSSN 7    MSSN 8    MSSN 9    MSSN 10,2X,18H/    EFF        PCEF
     2F  )
      DO 22 K=1,J
      WRITE(6,211) K,(FOT(L,K),L=1,10),EFF(K),PCEFF(K)
  211 FORMAT(1HO,I3,10(4X,F5.4),4H /   ,F5.3,5X,F6.2)
   22 CONTINUE
C     WRITE OUT COST INPUT DATA
      WRITE(6,217)
  217 FORMAT(1H1,47X,24HCOST ANALYSIS INPUT DATA   )
      WRITE(6,213) COSLB,COSMAN,DEVEXP
  213 FORMAT(1H ,18HCOST/LB. DEPLOYED ,E10.4,24H DOLLARS ** COST/MANHR.
     1,E10.4,22H DOLLARS ** UNIT EXP. ,F5.3                              )
      WRITE (6,230)
  230 FORMAT(1HO,50X,20HNUMBER OF PROTOTYPES      )
      WRITE (6,231) (K,B(K),K=1,J        )
  231 FORMAT(1HO, 10(I2,3X,F4.0,2X))
      WRITE(6,214)
  214 FORMAT(1HO,40X,38HSYSTEM AND SUBSYSTEM WEIGHTS IN POUNDS   )
      WRITE(6,215)
  215 FORMAT(1H ,117HSYS        SUB 1        SUB 2        SUB 3        SUB 4
     1        SUB 5        SUB 6        SUB 7        SUB 8     /     TOTAL WT.
     2)
```

FIGURE D.1 -1 EFFECTIVENESS - COST COMPUTER PROGRAM
(CONTINUED)

```
      DO 24 K=1,J
      WRITE(6,216) K,(SUBWT(L,K),L=1,8),WEIGHT(K)
  216 FORMAT(1H0,I3,8(6X,F6.1),11H      /         ,F6.1)
   24 CONTINUE
      WRITE(6,218)
  218 FORMAT(1H0,42X,35HWT OF CONSUMABLES (LB) PER MISSION   )
      WRITE(6,204)
      DO 25 K=1,J
      WRITE(6,219)  K,(CONSUM(L,K),L=1,10)
  219 FORMAT(1H0,I3,10(3X,F6.1))
   25 CONTINUE
      WRITE(6,220)
  220 FORMAT(1H0,41X,36HASTRONAUT TIME PER MISSION IN HOURS    )
      WRITE(6,204)
      DO 26 K=1,J
      WRITE(6,221)  K,(ASTIME(L,K),L=1,10)
  221 FORMAT(1H0,I3,10(3X,F6.1))
   26 CONTINUE
      WRITE(6,226)
  226 FORMAT(1H0,36X,28HCOST OF SUBSYSTEMS - DOLLARS    )
      WRITE(6,227)
  227 FORMAT(1X,113HSYS       SUB 1          SUB 2          SUB 3          S
     1UB 4          SUB 5          SUB 6          SUB 7          SUB 8    )
      DO 28 K=1,J
      WRITE(6,228) K,(SUBCOS(L,K),L=1,8)
  228 FORMAT(1H0,I3,8(4X,E10.4))
   28 CONTINUE
C     WRITE OUT COST RESULTS
```

FIGURE D.1-1 EFFECTIVENESS - COST COMPUTER PROGRAM (CONTINUED)

```
      WRITE(6,222)
  222 FORMAT(1H1, 37X,44HOVERALL COST ANALYSIS IN DOLLARS PER SYSTEM   )
      WRITE(6,223)
  223 FORMAT(1H ,117HSYS        NO.SYS       CONSUM      FIXED       TOTAL
     1  VEHICLE OP.             R+D                CREW OP.          TOTAL OP.
     2   )
      WRITE(6,224)
  224 FORMAT(1H ,119H             USED        WT.-LB      WT.-LB      WT.-LB
     1  COST-DOLLARS        COST-DOLLARS        COST-DOLLARS        COST-DOLLA
     2RS   )
      DO 27 K=1,J
      WRITE(6,225) K,DEPLOY(K),CWT(K),FWT(K),TWT(K),CSTOP(K),DEVCST(K)
     1,CSTMAN(K),TCOST(K)
  225 FORMAT(1H0,I3,5X,F5.1,1X,3(3X,F8.0),4(6X,E12.4))
   27 CONTINUE
C     TESTS FOR MORE TRIALS
      IF(NRUN.GT.JO) GO TO 1
      CALL EXIT
      END
```

FIGURE D.1-1 EFFECTIVENESS - COST COMPUTER PROGRAM (CONCLUDED)

CARD 2

cc 1-3-(I3) Number of Systems

cc 4-6-(I3) Number of Types of Missions (maximum for the present = 10.)

cc 7-9-(I3) Number of Subsystems per System (maximum for the present = 8)

cc 10-19 (E 10.4) Cost/LB to Deploy System on moon

cc 20-29 (E 10.4) Cost/Man Hour of Lunar Stay

cc 30-34 (F 5.4) Development Exponential

CARD 3 (This may require more than one card)

cc 1-80 (F4.1) Input Potentials of All Systems to do Various missions. Read in the following order:

SYS 1, Mission 1, SYS 2, Mission 2, -- SYS 1, Mission N

SYS 2, Mission 1 -- SYS 2, Mission N -- SYS J, Mission N

Thus, if you have 5 systems and 10 missions, you require 100 spaces, therefore, one puts 80 on one card and the first 20 on the next. (40 entries per card).

CARD 4 (May need more than 1 card)

cc 1-80 (F 4.3) -- Write in all the Mission weighting factors in sequential order (80 columns per card or 20 entries)

Card 5 (May need more than 1 card)

cc 1-80 (F.4.1) Number of missions of each type (20 entries per card)

CARD 6 (May need more than 1 card)

cc 1-80 (F 5.0) Give total dry weight of each system in sequence. (16 entries per card)

CARD 7 (May need more than 1 card)

cc 1-80 (F 4.0) Number of vehicles of each system type in sequence (20 entries per card)

CARD 8 (May need more than 1 card)

cc 1-80 (F 5.0) Weight of each subsystem for SUB 2--SYS 1, SUB NO, SYS 2, SUB 1--SYS 2, SUB NO--SYS J, SUB NO. (16 entries/ card)

CARD 9 (May need more than 1 card)

cc 1-80 (F 5.1) Give Time (hours) which an astronaut must spend on each mission sequence -- same as Card 3. (16 entires/card)

CARD 11 (May need more than 1 card)

cc 1-80 (E 10.4) Cost (Dollars) each subsystem for all systems in sequence given in card 8. (8 entires/card)

CARD 12 (May require more than 1 card)

cc 1-80 (F 5.0) Number of prototypes of each system type taken in sequence (16 entires per card)

CARDS 2-11 are repeated the number of times specified in CARD 1. This allows one to try a series of parameters on all systems on the same run.

D.3 PROGRAM OUTPUT

A complete set of information is given as output from this program. This includes all the input information, weighted effectiveness results, total effectiveness, percent effectiveness and cost figures for operation research and development and crew operations. Finally the total costs are given.

APPENDIX F

PHILOSOPHY OF METHODS

R. Ray Nachlinger

This Appendix is devoted to a general discussion as to why the methods of Part II were used in our evaluation. Since the result of any evaluation is a decision, we begin by discussing abstractly. We then describe our general procedures and how they fit into the abstract framework.

F.1 Decisions

When one is asked to make a decision, he is actually asked to answer the question: Which of two alternatives will produce the more desirable result? The answer to this question, however, involves two considerations: (1) what results will the various alternatives produce, and (2) what does one mean by better? Since a decision becomes purely mechanical once these two questions have been successfully answered, we see that the difficulty of a decision is in obtaining answers to these questions. We will thus direct our attention to them. Of these two questions, the first is perhaps the most trouble-some operationally. For this reason, it normally receives the most attention, sometimes to the point where the second question is not considered at all. Another reason that the first question receives so much attention from scientists and engineers is that this question is much more amenable to mathematical modeling and quantitative analysis than the second. These quantitatively oriented people thus fool themselves into thinking that they have made an objective decision, when, in fact, all they have done is generate the probable results of alternatives. While we will readily admit that the generation of reliable results is difficult and of importance, it is more straightforward than the second question, and we will, therefore, consider the second question in greater depth.

Any alternative will, in general, result in three classes of results: good, bad, and indifferent. Thus the problem, which appeared simple, to decide which of two alternatives is the better, takes on another dimension. One must not only know what is "good", but he must also know how much bad he is willing to tolerate to obtain a certain amount of "good". It, by now, has become obvious that in all but the simplest decision, one person could not possibly keep all these factors straight, perform all the work, and obtain a rational decision if he did not have a systematic procedure. There have been many such procedures developed that vary in detail, but they are all essentially the same.

The main objective of any of these systematic procedures is the presentation of the good vs bad results in a form which will facilitate making the decision as to which is better. A secondary goal of these procedures is to document how these data were obtained. This facilitates checking and changes. It should be noticed that <u>none</u> of these procedures <u>makes</u> a decision, they only generate and present information that facilitates the making of a decision. One should also notice that use of any of these procedures assumes one already knows not only what is good and bad, but that he can also order results as to degree.

We will now turn to the most important factor in making a decision; how do we decide good, bad, and better. Before proceeding further, it should be noticed that we are not using bad as the opposite of good. Here, bad refers to an undesirable side effect, e.g., while one might find an around-the-world trip highly desirable, the time required would most likely be a bad side effect. If we consult the dictionary for the definition of good, we find that something is good if it is sufficient or adequate for our purposes. Thus, we see that to answer whether or not something is good, we must first know our purposes. This fact, that before one can intelligently make a decision he must first have a defined objective, is the most overlooked fact in writings on the systematic procedures.

We now have a way of defining good: A result is good if it contributes to the attainment of a goal or objective. We can also order good results by asking which one contributes more toward the objective. In fact, we can associate a goodness number with each result by considering the percentage of the objective attained by that alternative. i.e. Zero if nothing is accomplished, 100 if all the goal is attained. We will define a badness number as a measure of our resources expended by an alternative that could be used for other objectives.

With these definitions of good and bad, given any objective and a set of alternatives, we could go through a systematic procedure and obtain a graph of the good vs the bad for each alternative. The question now becomes: Which alternative is the best? The answer to this question also lies with an objective, but this objective is not the same one from which we decided good. The objective that we must consider now is some overall objective of which our original objective is a sub-objective. Only by considering how our sub-objective fits into the overall picture can we make decisions such as how much are we willing to give up of something else to obtain our goal.

By now, it should be obvious that no decision can be objective, since all decisions have parts that must be decided by value judgments. In fact, the only part of the above process that can be made objective is the calculation of the results of alternatives. This portion of objectivity is sometimes unfortunate, since some people, in their vain quest for objectivity, confuse this one portion with the whole decision making process. This makes them believe their numbers are an infallible decision, forgetting all the subjectivity that is involved.

In summary, we see that in order to make a decision, one must consider the following: (1) an objective, (2) a major goal, (3) an idea of how important the objective is with respect to other objectives which form the goal, (4) a set of alternatives, and finally, (5) the results of each alternative.

F.2. Evaluation of Mobilities Systems

In keeping with the abstract discussion, the first task that was attacked was the establishing of objectives and rating them in importance within an overall goal. The objectives that were defined were the nominal missions discussed in Chapter 5. The importance of each mission was assessed within the overall goal of lunar exploration. These definitions provided the framework within which all successive decisions were made.

When this was completed, it became obvious that we were faced with two types of decisions. First, we must decide what general class of mobility device we should use, and second, we must decide what subsystems would yield the best vehicle of the type chosen. Since these two types of decision are quite different, different systematic procedures were developed for each type.

The first decision was so important that two procedures were developed, and applied so that a check could be made. The first method used was to measure a candidates ability to perform the missions vs the amount of resources expended in accomplishing that mission, and ranking the candidates by these factors. Since the results of the two procedures agreed, we decided to consider only near-ground vehicles in the second phase. The details of each of the above analyses can be found in other sections of this report.

We would like to point out that there is another aspect to the problem that we could not consider because of time constraints.

A decision must also be made as to how many vehicles should be obtained to perform the program. It is also reasonable to ask whether or not it would be desirable to have more than one type of vehicle, and if so how many types and how many of each. The first type of analysis could be easily performed to answer these questions.

Since the second type of decision is quite different than the first, a third procedure was developed. At this stage, the gross requirements were already defined, so that what remained was to develop the best configuration that met these requirements. To accomplish this, attention was turned to the subsystems. The best subsystem for each task was decided, and a configuration was assembled from these. This procedure is discussed in Chapter 10.

APPENDIX G

CONSIDERED ACRONYMS

ACRONYM	NAME	COMMENTS
LUV	LUNAR VEHICLE	TOO GROOVY
MOVE	MOBILITY VEHICLE	NOT SURE THAT IT WOULD MOVE
PLUMS	PLANETARY LUNAR MOBILITY SYSTEM	A PERISHABLE
LTV	LUNAR TRANSPORT VEHICLE	SAME AS LING-TEMCO-VOUGHT
LSD	LUNAR SURFACE DEVICE	UNLAWFUL
MUMS	MANNED UNMANNED MOBILITY SYSTEM	TOO MUCH FLOWER POWER
LRS	LUNAR ROVER SYSTEM	SIMILAR TO IRS
LUC	LUNAR CAR	CAR DENOTES POLLUTION
PLASMA	PLANETARY SURFACE MOBILITY AID	TOO BLOODY
MOUSE	MANNED OR UNMANNED SURFACE EXPLORER	NOT CERTAIN THAT MOON IS GREEN CHEESE
MMMMMM	MANNED MECHANISM FOR MOBILITY MISSION ON MOON AND MARS	TOO SUGGESTIVE
SMUT	SYSTEM FOR MAN OR UN-MANNED TRANSPORTATION	SUPREME COURT RULING NOT CLEAR
SLUT	SYSTEM FOR LUNAR UNMANNED TRANSPORT	NO GOOD IN THE UNMANNED MODE